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# **NASA**

## **ORBIT TRANSFER ROCKET ENGINE TECHNOLOGY PROGRAM**

### **FINAL REPORT**

**Prepared Under  
Contract NAS3-23171  
for  
National Aeronautics and Space Administration  
Lewis Research Center  
21000 Brookpark Road  
Cleveland, Ohio 44135**

**Prepared by  
Pratt & Whitney Aircraft  
Government Products Division  
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## **FOREWORD**

This report presents the results of the Orbit Transfer Rocket Engine Technology Program. The study was conducted by Pratt & Whitney Aircraft, Government Products Division (P&WA/GPD) of the United Technologies Corporation (UTC) for the National Aeronautics and Space Administration Lewis Research Center (NASA/LeRC) under contract NAS3-23171

This study was initiated in November 1981 with the technical effort completed in 15 months. The study effort was conducted under the direction of the LeRC Space Technology Directorate with Dr. Larry P. Cooper as Contracting Officer's Representative. The effort at P&WA/GPD was carried out under the direction of Mr. James R. Brown, Program Manager.

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## **SECTION 1.0 INTRODUCTION**

### **1.1 OBJECTIVE**

The objective of the Orbit Transfer Rocket Engine Technology Program was to provide conceptual designs, and identify, screen, and propose advanced technology concepts which would benefit future hydrogen/oxygen Orbit Transfer Vehicle (OTV) propulsion systems

### **1.2 BACKGROUND**

The National Aeronautics and Space Administration — Office of Aeronautical Space Technology (NASA-OAST) long-range plan includes technology efforts for high performance hydrogen/oxygen rocket engines for use in future OTV propulsion systems. The subject of future OTV propulsion has been addressed since the early 1970's and has included efforts to define candidate engine systems and identify their technology requirements. Recent studies have focused on the expander cycle concept with emphasis on maximizing performance.

Various research and technology efforts during the 1970's produced concepts and demonstrated capabilities that were incorporated into the Advanced Expander Cycle Engine design. Similarly, this study and subsequent research and technology should provide advanced capabilities that can be incorporated into an OTV engine whose development would begin in 1990.

### **1.3 DOCUMENTATION OF RESULTS**

This report presents the results of the study which are detailed in the following sections. The study logic flow and task description is presented in Section 2.0 and a description of the study starting point engine design (the 1980 OTV AEE) is given in Section 3.0. Advanced technology concepts are given in Sections 4.0 (turbomachinery), 5.0 (thrust chamber/nozzle), and 6.0 (throttling). Section 7.0 gives the evaluation and ranking of the technology concepts formulated in this study. Section 8.0 presents a description of the 1990 technology advanced OTV engine. Conclusions and recommendations are presented in Section 9.0.

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## SECTION 2.0

### STUDY LOGIC AND TASK DESCRIPTION

The Orbit Transfer Rocket Engine Technology Program addressed the use of advances in technology to enhance and improve the capabilities of hydrogen/oxygen engine for an OTV application. The engine requirements and capability goals are given below.

#### *Requirements*

- Nominal thrust level — 10,000 to 25,000 lb
- Mixture Ratio —  $6.1 \pm 1.0$
- Gimbal Angle —  $\pm 6$  deg
- Propulsive use of chutdown propellants

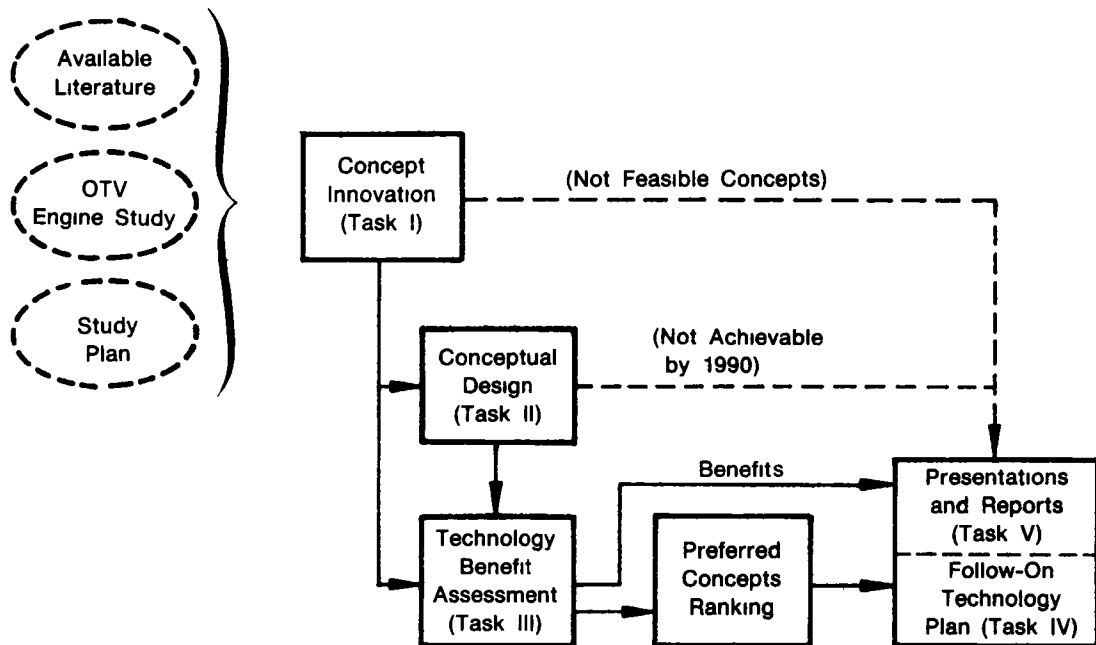
#### *Goals*

- Vacuum Specific Impulse — 520 sec
- Vacuum Thrust Throttle Range — 30:1
- Net Positive Suction Head (NPSH) ( $H_2$  and  $O_2$ ) — 0
- Weight — 360 lb (max)
- Length (stored) — 40 in (max)
- Reliability — 1.0
- Service Life Between Overhauls — 500 starts/20 hr (min)
- Service Free Life — 100 starts/4 hr (min)

The program was divided into five tasks and the relationship of these tasks is shown in Figure 2-1. The Task I activity centered on the initial formulation of concepts for the advanced engine. Subcomponent, component, and engine concepts were formulated with the intent of maximizing expander cycle engine capabilities in view of the basic requirements and goals listed below. An important factor considered at the subcomponent and component level was the utilization of advanced materials. The formulated concepts were screened and those which were not feasible (i.e., required a technology breakthrough rather than an advance in the current state-of-the-art) were eliminated. Those concepts that remained were carried over into Task II.

A preliminary design of each of the concepts was generated in Task II to determine characteristics for engine cycle analysis. Advanced expander engine cycles were then synthesized to maximize engine capability utilizing these new concepts. These cycle optimizations were conducted at the selected baseline of 15,000 lb thrust, a mixture ratio of 6.1 and an engine installed length of 40 in. The cycle studies were conducted using the Pratt & Whitney advanced expander cycle point design computer program. This program provides the capability to reoptimize the engine cycle as individual component and/or cycle configuration characteristics are defined by design analysis.

The advanced features were also evaluated to estimate the time required to bring these features within the industrial technology base. Those features estimated not within the constraint of 1990 technology (i.e., could not reasonably be expected to be technologically ready by 1990 even with a focused technology program) were eliminated from further consideration. Cycles using advanced features, which were estimated to be within the required technology base, were examined in detail. The advanced features of these cycles were then ranked, in conjunction with the Task III effort, considering the potential gain in engine capability produced by the feature.



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Figure 2-1 — Study Task Relationship

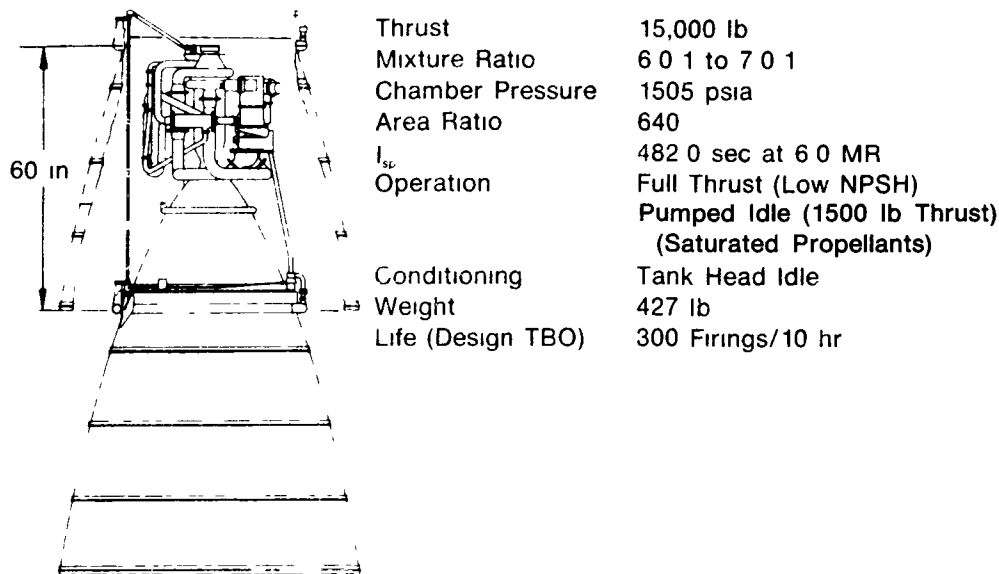
A methodology was developed under the Task III effort to assess the relative benefits of the feasible concepts. Engine capability improvement was evaluated using this methodology relative to the baseline 1980 advanced expander cycle engine (AEE) as well as a relative ranking of the RL10A-3-3 engine, and the capability goals of the Statement of Work (SOW) for the 1990 AEE.

Based on the benefit assessment evaluation, a preferred technology concept ranking list was generated, and a technology acquisition plan formulated under Task IV. The individual plans themselves were ranked considering engine system capability improvements, costs, probability of successful acquisition of the technology within the required time frame (prior to 1990), and general applicability to the selected OTV engine system concept. The Task V activity was the generation of the study documentation other than the technology plan.

### SECTION 3.0 BASELINE ENGINE DESCRIPTION

The advanced expander cycle engine (AEE), shown in Figure 3-1, was used as the baseline for this study. This clean sheet advanced-technology engine is described in this section. Basically it was a 1980 state-of-the-art design optimized specifically for use in the man-rated OTV. The baseline AEE had the following requirements:

- 1 Thrust level 15,000 lb at 6:1 mixture ratio
- 2 Performance optimized
- 3 Operating modes
  - Tank head idle
  - Pumped idle
  - Low NPSH pumping capability at full thrust
- 4 Design life 300 firings and 10 hr



FD 219109

Figure 3-1 — Baseline Engine

The design of the AEE was the result of NASA funded studies carried out during the 1970's culminating in the "Advanced Expander Cycle Engine Point Design Study" (contract NAS8-33567, final report dated 15 March 1981).

High engine performance, packaged within a given envelope, is an important factor in most OTV applications. Selection and design of each engine component must be evaluated to ensure successful operation within its particular envelope, while delivering optimum performance. The AEE takes a significant step toward the maximum achievable specific impulse with a chamber pressure of 1500 psia and with the incorporation of a high area ratio nozzle. (The currently operational  $H_2/O_2$  space engines have chamber pressures in the 400-500 psia range and area ratios of approximately 60.)

As chamber pressure is increased and the 15,000 lb thrust level is held constant, throat area decreases, reducing nozzle exit diameter and length for a given area ratio. Conversely the high  $P_c$  allows a significantly increased nozzle area ratio within the same envelope (from approximately 60 to over 600) which increases the specific impulse on the order of 40 seconds. In addition, this increase in nozzle area ratio does not have a significant impact on engine weight due to the decreased nozzle surface area for a given area ratio.

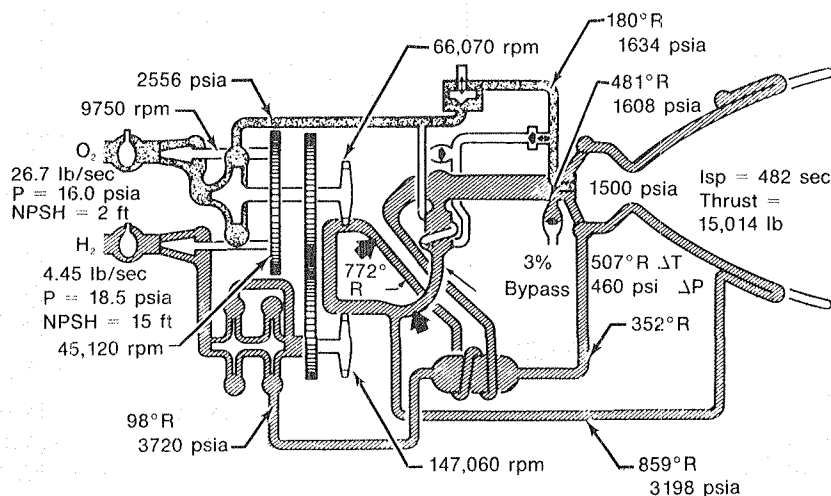
The increased chamber pressure of the AEE is due to the use of a preheater in the power cycle, improved technology turbopumps with higher efficiencies, an improved heat transfer combustion chamber, and reduced engine power margin.

The preheater (regenerator) allows chamber pressure to be increased by preheating the chamber coolant with the turbine discharge flow. This raises turbine inlet temperature, and increases turbine fluid available power. Additional heat for the cycle is supplied by a thrust chamber optimized to maximize coolant exit temperature.

While it is obvious that high specific impulse is valuable, quantification is more difficult. For example, the payload sensitivity to changes in Isp may range from 40 to 85 lb payload/sec Isp. Various estimates have been made for the cost/lb of payload delivered to GEO but these generally fall in the range of 6,000 to 13,000 \$/lb. If a model of 100 missions is used and it is assumed that an average of 100 percent of the additional payload capability due to an increase in specific impulse is utilized per flight, then for each second of increase in specific impulse, a total program savings of between \$24M and \$111M will be realized. This analysis does not account for the cost of engine development. In all probability, an advanced engine will not be solely justified on the basis of increased Isp but rather on the basis of an operational requirement(s) that cannot be met with existing engines complimented with increased performance.

### 3.1 ENGINE STEADY-STATE CHARACTERISTICS

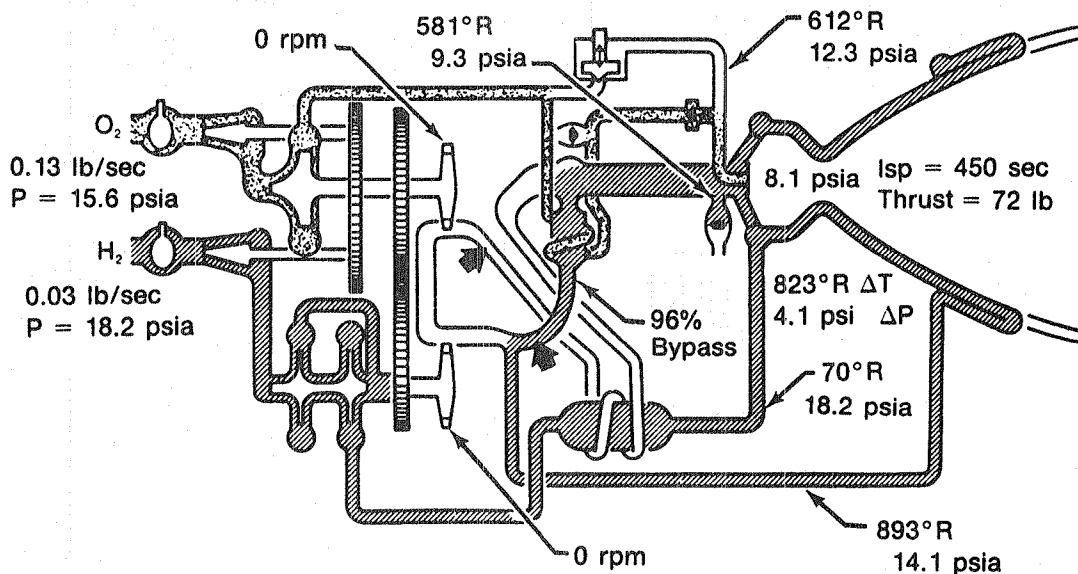
The steady-state cycle parameters of the AEE are presented in Figures 3-2 through 3-5 for the operating points of full thrust (O/F = 6:1 and 7:1), pumped idle, and tank head idle.



FD 219104

Figure 3-2. — Advanced Expander Engine Propellant Flow Schematic at Full Thrust (O/F = 6.0)





FD 219107

Figure 3-5. — Advanced Expander Engine Propellant Flow Schematic at Tank Head Idle (O/F = 4.0)

A brief description of the propellant paths at the engine design point (100 percent thrust and O/F = 6:1) shown in Figure 3-1 follows. Fuel (hydrogen) enters the engine through a ball-type inlet shutoff valve mounted on the inlet of a low-pressure pump (boost pump) that is gear-driven from the main oxidizer turbopump shaft. The low-pressure pump operates at a rotational speed of 45,100 rpm with a 15 ft NPSH capability. From the low-pressure pump, fuel enters the first of two back-to-back shrouded impeller centrifugal stages. The impellers are mounted on a shaft driven by a single-stage, low reaction, full admission turbine. The high-pressure pump operates at the nominal speed of 147,100 rpm. Approximately 5.8 percent of the fuel is used as a thrust-piston balancing flow for the high-pressure pump. This flow is taken off at the second-stage discharge, circulated to the thrust piston, and then injected back into the propellant flowpath at the high-pressure pump interstage.

The fuel moves from the high-pressure pump discharge and enters the hydrogen-hydrogen regenerator which utilizes energy from the turbine discharge flow to preheat the chamber coolant. The regenerator is a cross-flow heat exchanger which increases the temperature to approximately 350°R, providing the fuel in a gaseous state for cooling the thrust chamber. The chamber regenerative coolant enters an inlet manifold located at the injector face plane and flows into and through the nontubular (copper alloy liner and electroformed nickel shell) combustion chamber downstream past the throat to an area ratio of approximately 6:1. There the coolant enters the tubular nozzle section and flows down half of the tubes to an area ratio of 210:1 where a turnaround manifold routes it back (counter to the combustion gas flow) through the remaining tubes. At an area ratio of approximately 60:1 where the double pass construction starts, the flow is collected in a manifold and is withdrawn.

At the jacket discharge, the fuel flow is split, with about 3 percent of the flow bypassed around the turbines. This flow passes through the GOX heat exchanger providing the heat transfer for gaseous oxygen tank pressurization capability, if required. The remaining 97 percent of the flow is routed through the turbines to provide the power to drive the turbopumps, and

then through the hot side of the hydrogen-hydrogen regenerator. After leaving the regenerator, the turbine bypass flow re-enters the mainstream and hydrogen tank pressurization flow is removed through the tank pressurization supply connection, if required. The flow is then injected into the thrust chamber.

Oxidizer (oxygen) enters the engine through an inlet valve similar to the fuel-side inlet valve. A low-pressure oxidizer pump, geared from the main oxidizer turbopump and operating at a shaft speed of 9,750 rpm, provides the engine with a 2 ft NPSH capability. The discharge from the low-pressure pump enters a single-stage, shrouded, centrifugal-type, high-pressure pump driven at a speed of 66,100 rpm by a single-stage, low reaction, full admission turbine. Oxidizer tank pressurization, if required, is taken off downstream of the pump through a heat exchanger where it is vaporized by hot fuel, and, is routed through the oxidizer tank pressurization connection to the vehicle tank. The remainder of the flow continues to the oxidizer control valve, which is preset to give the desired mixture ratio. From the control valve, the flow enters the injector manifold and is injected into the combustion chamber.

A hydrogen-oxygen torch igniter is used to light the main combustion chamber. Fuel for the igniter is tapped off immediately downstream of the turbines, and gaseous oxidizer is supplied from the tank pressurization GOX heat exchanger.

During pumped idle operation (Figure 3-4), thrust is set at approximately 10 percent of the rated level. This is accomplished by bypassing 54 percent of the total fuel flow around the turbine. The increased turbine bypass flow also serves the purpose of providing the energy to the oxygen which is diverted around the oxidizer control valve to a heat exchanger. This delivers gaseous oxygen to the injector, resulting in greater combustion stability at the reduced pressure levels. At tank head idle (Figure 3-4), which is utilized for pump cooldown and propellant settling, the pumps and turbines do not rotate. The fuel flow bleeds down through the pumps, regenerator, and jacket where it enters the turbine bypass leg. Here the flow splits with approximately 10 percent being routed to the hot side of the regenerator to provide energy to the cold side, keeping vapor at the jacket inlet. The remaining flow goes through the heat exchangers, vaporizing the oxidizer flow. This results in a thrust level of approximately 70 lb.

Simple open-loop control of the engine assures stability. Stable control operation at the three thrust levels is achieved by time sequencing five solenoid valves which pressurize main valve cavities to establish the proper valve positions at each thrust setting. Ground mixture ratio adjustment at each of the three thrust settings is provided.

Two of the valves have pressure feedback during the transition between thrust settings, yet the valve positions are hard against a stop during steady-state operation. Should loss of electrical power or helium pressure occur, all valves will move to their fail-safe position and a safe engine shutdown will result.

The engine is transitioned from one thrust setting to another utilizing vehicle electrical signals. A schematic showing the location of each valve is provided in Figure 3-6. The five solenoid valves respond to the electrical signals by opening the appropriate valve cavity to a pressure source, either helium, hydrogen, or oxygen depending upon the application. These solenoid valves vent the valve cavities overboard when deactivated. Figure 3-7 shows typical operation and responding action by the engine control system.

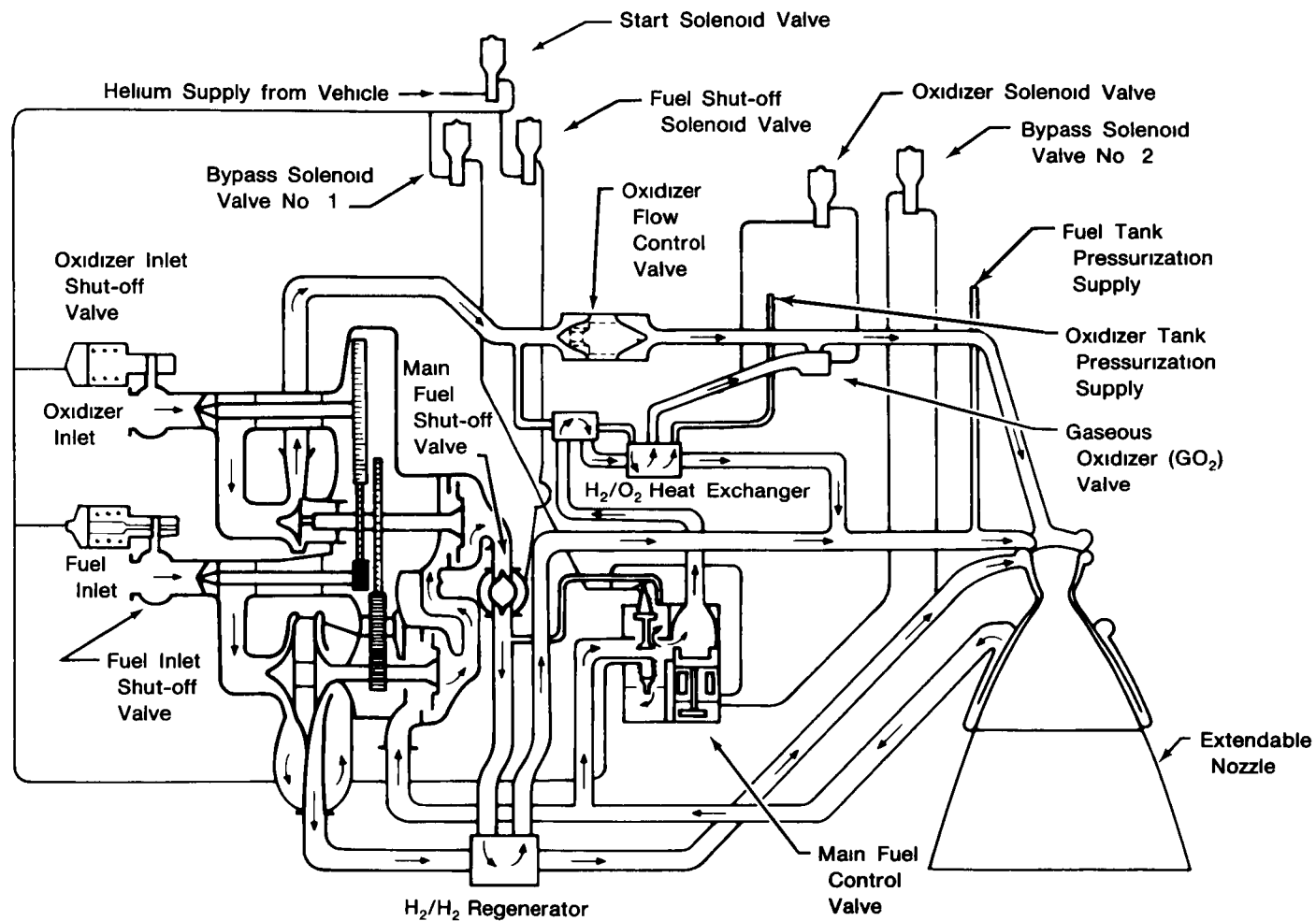


Figure 3-6 — Engine Control System Schematic

Start Solenoid Valve

Fuel Shutoff Solenoid Valve

Bypass Solenoid Valve No. 1

Bypass Solenoid Valve No. 2

Fuel and Oxidizer Inlet Valves

Main Fuel Shutoff Valve

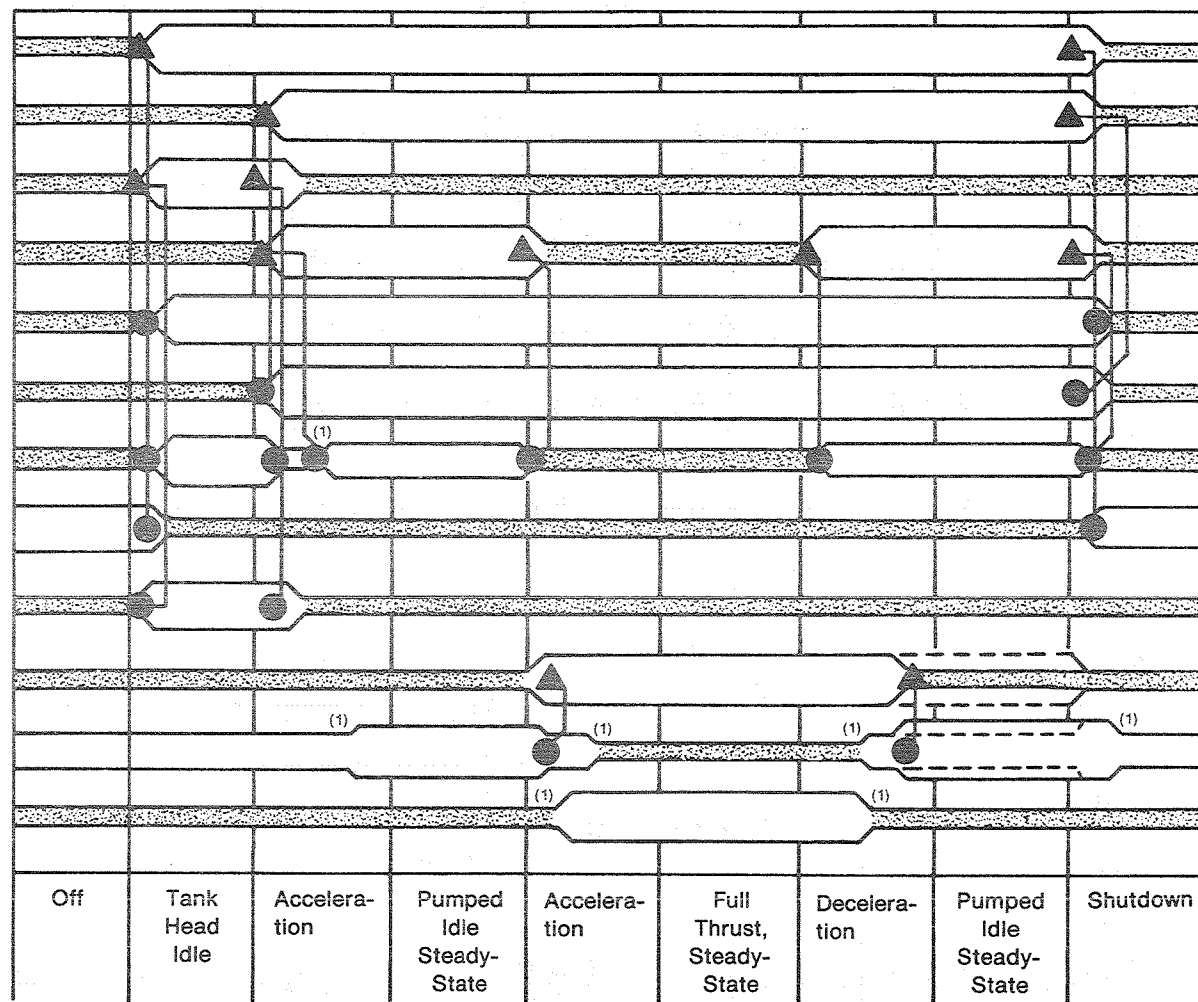
Main Fuel Control Valve

- Turbine Bypass Function
- Fuel Vent Function
- Regenerator Bleed

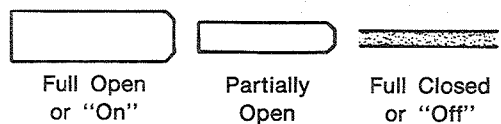
Oxidizer Solenoid Valve

Gaseous Oxidizer (GO<sub>2</sub>) Valve

Oxidizer Flow Control Valve



Legend:



▲ Items Marked ▲ Are Initiated by Command Signal and Cause the Successive Operations Marked ●

(1) Valve Actuated Automatically by Internal Engine Pressures

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Figure 3-7. — Value Sequence for a Typical Firing

Propellant shutoff is achieved using inlet shutoff valves which are low leakage cryogenic valves and are helium-actuated open during all phases of engine operation

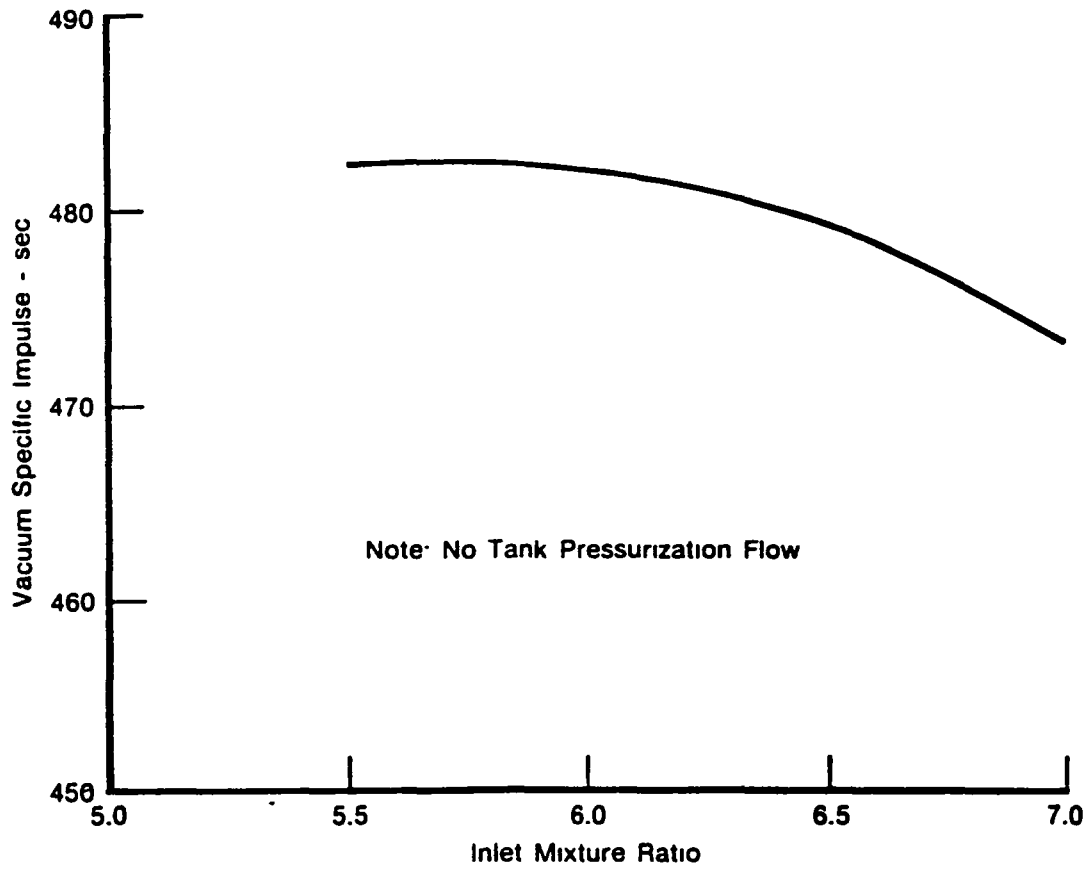
The main fuel shutoff valve is a low-pressure loss valve which is closed during the tank head idle mode of operation (zero speed) This valve is helium-actuated open during other phases of operation Shutdown is achieved by closing the main fuel shutoff valve as well as the main fuel control valve to starve the combustion chamber of fuel and cause flameout.

The oxidizer flow control valve is closed during the tank head idle and pumped idle modes It opens during the transition between pumped idle and full thrust when the oxidizer pump pressure rise is above 465 psid Ground adjustment of mixture ratio between 6:1 and 7.1 is provided at the full thrust setting

The gaseous oxidizer ( $\text{GO}_2$ ) valve provides two functions The first function is to allow mixture ratio to change from 4:1 at tank head idle to 6.1 at pumped idle The second function is to change the phase at the oxidizer injector from gas to liquid as the engine accelerates from pumped idle to full thrust Ground adjustment of mixture ratio at tank head idle and pumped idle is provided

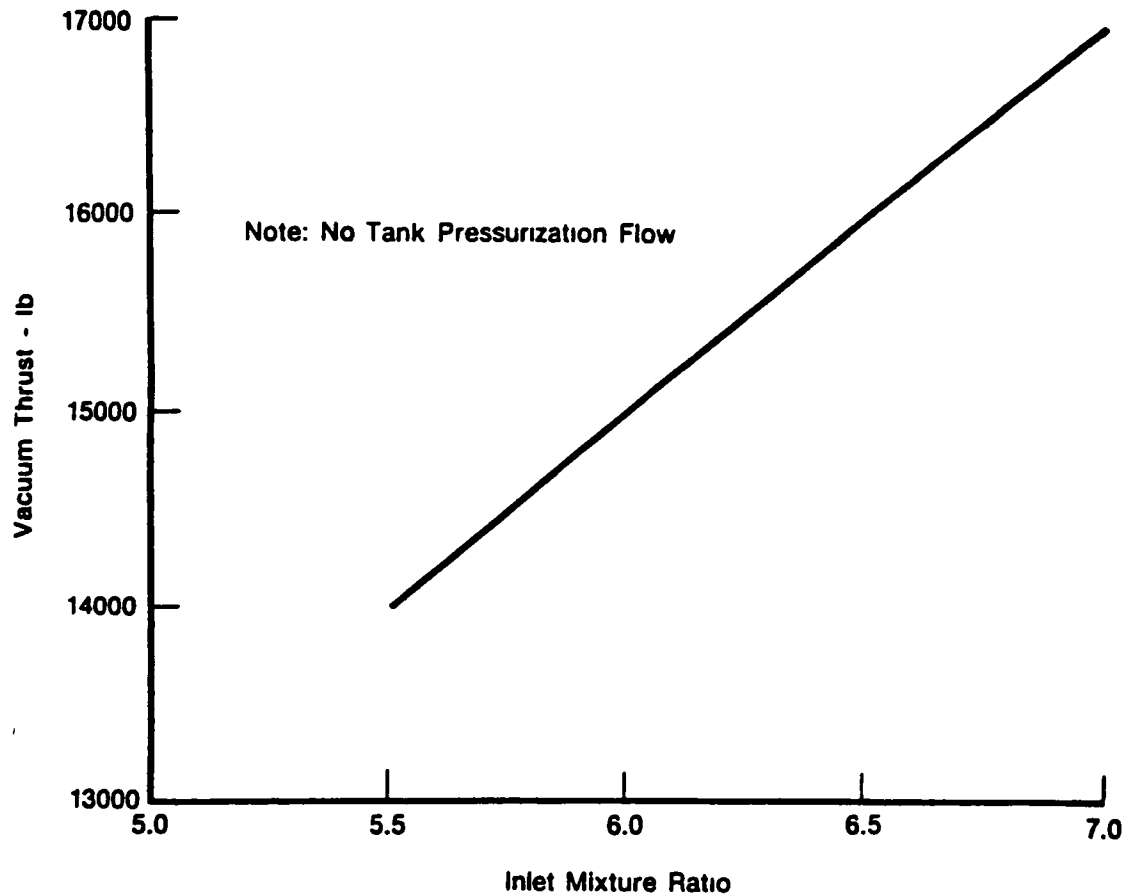
The main fuel control valve has three functions One function is to vent fuel overboard during shutdown, a second function is to direct flow to the fuel regenerator hot side during tank head idle, and the third function is to set turbine bypass flow during the three thrust settings Ground adjustment of this valve at each of the three thrust levels is accomplished by adjusting the needle valve at the full thrust level and the stop positions of the valve at the tank head idle and pumped idle thrust levels

Off-design full-thrust specific impulse and thrust characteristics are presented in Figures 3-8 and 3-9 respectively The estimated life of the engine at the various operating points is shown in Table 3-1 and the estimated weight of the engine and its various components is given in Table 3-2



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Figure 3-8 *Estimated Effect of Inlet Ratio on Vacuum Specific Impulse at Full Thrust*



FD 212853

Figure 3-9 Estimated Effect of Inlet Mixture Ratio on Vacuum Thrust at Full Thrust Setting

TABLE 3-1 — ESTIMATED ADVANCED EXPANDER CYCLE ENGINE LIFE

| Operating Point       | Time Between Major Overhauls <sup>(1)</sup> |                  |
|-----------------------|---|------------------|
|                       | (Cycles <sup>(2)</sup> )                    | (hr)             |
| Full Thrust (O/F=6 1) | 1500  | 10               |
| Full Thrust (O/F=7 1) | 650   | 10               |
| Pumped Idle           | >2000                                       | >30              |
| Tank Head Idle        | >2000                                       | (Not Applicable) |
| Design Goal           | 300   | 10               |

Notes

(1) Operation without major component changes (e.g., thrust chamber/primary nozzle, turbopump)

(2) A cycle is defined as an engine thermal cycle up to the indicated thrust level (e.g., tank head idle to pumped idle to full thrust (O/F=6 1) to pumped idle to shutdown would be one full thrust (O/F=6 1) cycle)

TABLE 3-2 — ESTIMATED ADVANCED EXPANDER CYCLE  
ENGINE WEIGHT

| <i>Item</i>                                     | <i>Material</i>                               | <i>Weight, lb</i> |
|---|---|-------------------|
| <i>Primary Nozzle Assy</i>                      |   |                   |
| Cooling Tubes                                   | 347 SST                                       | 31 0              |
| Thrust Chamber/Injector                         | 347 SST, N-155 Rigmesh, Amzirc                | 58 1              |
| Primary to Secondary Seal                       | 347 SST                                       | 12 0              |
| <i>Secondary Nozzle Assy</i>                    |   |                   |
| Nozzle Shell                                    | Uncoated Carbon/Carbon                        | 60 2              |
| Nozzle Supports                                 | Uncoated Carbon/Carbon                        | 8 8               |
| <i>Screw Jacks and Actuation</i>                |   |                   |
| Screw Jacks                                     | Uncoated Carbon/Carbon                        | 7 7               |
| Bearings and Housings                           | 347 SST                                       | 6 9               |
| Gear Drive and Drive Motor                      | 347 SST                                       | 5 9               |
| <i>Gimbal Mount</i>                             | Aluminum Alloy                                | 4 0               |
| <i>Turbopump Assy</i>                           | Al Alloy, 347 SST, 17-7 PH,<br>A-286 Titanium | 60 7              |
| <i>Heat Exchangers</i>                          |   |                   |
| H <sub>2</sub> Regenerator                      | Aluminum Alloy                                | 32 8              |
| Vortex Pre vaporizer                            | Aluminum Alloy                                | 5 2               |
| GOX Heat Exchanger                              | Aluminum Alloy                                | 16 3              |
| <i>Control Valves</i>                           | Al Alloy, 347 SST, 17-7 PH,<br>A-286          | 54 0              |
| <i>Miscellaneous</i>                            |   |                   |
| (Plumbing, Solenoids,<br>Instrumentation, etc ) |   | 63 0              |
| <b>Total</b>                                    |   | <b>426 6</b>      |

## **3.2 ENGINE TRANSIENT CHARACTERISTICS**

### **3.2.1 Ignition**

Ignition occurs during the first 0.25 sec of the tank head idle transient. The tank head idle mode is used to condition the pumps prior to rotation.

The start solenoid valve and the bypass solenoid valve No. 1 are energized causing the fuel and oxidizer inlet shutoff valves, the turbine bypass, and the fuel regenerator poppet to open and causing the fuel vent poppet to close. Spark to the torch igniter is initiated immediately and terminated once the torch lights (about 0.2 sec).

Chamber ignition is approached from the oxidizer rich side, the oxidizer side fills more rapidly than the fuel because of its reduced volume which allows simultaneous opening of the fuel and oxidizer inlet shutoff valves. A torch igniter is used to provide the ignition energy.

### **3.2.2 Pump Conditioning**

Pump conditioning is implemented utilizing the tank head idle mode where the fuel bypasses the turbine and the pumps do not rotate. About 2 min are required to condition the pump housings and impellers from an initial temperature of 500°R to the temperature level of the propellants in the tank during which time 4 lb of fuel and 16 lb of oxygen are consumed. This represents less than 1 sec of full thrust consumption with a specific impulse penalty of only 7 percent.

### **3.2.3 Tank Head Idle to Pumped Idle**

Once the pump housings and impellers are cooled to tank temperatures during the tank head idle mode, the engine may be transitioned to the pumped idle thrust setting. The bypass solenoid valve No. 1 is closed, bypass solenoid valve No. 2 is opened, and the fuel shutoff solenoid valve is opened. Closing the bypass solenoid valve No. 1 closes the turbine bypass valve and closes the bleed valve supplying flow to the regenerator hot side. With the main fuel shutoff valve open and the turbine bypass valve closed, all the fuel is directed through the turbine producing the maximum available torque for breakaway. Opening bypass solenoid valve No. 2 allows the bypass valve to open to its pumped idle setting as valve inlet pressure (speed) increases. Also as speed increases, the gaseous oxidizer valve opens further to adjust mixture ratio from 4:1 at tank head idle to 6:1 at pumped idle.

### **3.2.4 Pumped Idle to Full Thrust**

The transition from the pumped idle thrust setting of 1500 lb to the full thrust setting of 15,000 lb is initiated by closing the bypass solenoid valve No. 2 and opening the oxidizer solenoid valve. Closing the bypass solenoid valve vents the turbine bypass valve cavity causing the bypass valve to close. Opening the oxidizer solenoid valve allows the gaseous oxidizer valve to close as valve inlet pressure increases. The oxidizer flow control valve will open when the overall oxidizer pump pressure rise is greater than 465 psid. The opening of the oxidizer flow control valve occurs just prior to the gaseous oxidizer valve closure. The pressure level for the switch from gaseous oxidizer at the injector to liquid is chosen so that the injector pressure differential ( $\Delta P/P$ ) is sufficient to assure combustion stability.

### **3.2.5 Full Thrust to Pumped Idle**

For accurate shutdown impulse it was assumed that the engine would normally return to pumped idle prior to full engine shutdown.

The transition from full thrust to pumped idle is achieved by opening the bypass solenoid valve No 2. The bypass valve area opens as the valve cavity fills. Speed will decrease rapidly with the associated loss in turbine flow. The gaseous oxidizer valve will open when its upstream pressure drops below 600 psia followed by closure of the oxidizer flow control valve when the oxidizer pump pressure rise drops below 465 psid. The opening and closing of the two oxidizer valves directs the liquid oxygen through the oxidizer heat exchanger which changes it to a gas before it reaches the injector. The switch from liquid to gas is done to maintain sufficient injector velocity during low flow conditions to maintain stable combustion.

The oxidizer solenoid valve is closed 5 sec following the command to return to pumped idle. If shutdown is imminent the oxidizer solenoid valve may remain open allowing the shutdown signal to close it. However, if a steady-state pumped idle is required, the oxidizer solenoid valve should be closed anytime after the gaseous oxidizer valve opens (about 2 sec after the command is given to return to pumped idle).

### 3.2.6 Shutdown

As previously stated, it was assumed that the engine would normally be shut down from either pumped idle or tank head idle. Shutdown is initiated by removing voltage from all solenoid valves causing the other valves to move to their fail-safe positions. Both inlet shutoff valves close to eliminate further propellant consumption. A vent in the main fuel control valve opens to relieve pressure downstream of the fuel pumps. The fuel bleed valve, the turbine bypass valve (which is an integral part of the main fuel control valve), and the main fuel shutoff valve all close to starve the combustion chamber of fuel. The oxidizer control valve closes during the deceleration from full thrust when the oxidizer pump pressure rise drops below 465 psid. The gaseous oxidizer valve will open when power is removed from the solenoid. Oxidizer flow to the combustion chamber will continue until it is completely expelled downstream of the inlet shutoff valve. The fuel to the combustion chamber is depleted long before the oxidizer, causing an oxidizer rich flameout.

## 3.3 CRITICAL COMPONENT ANALYSIS

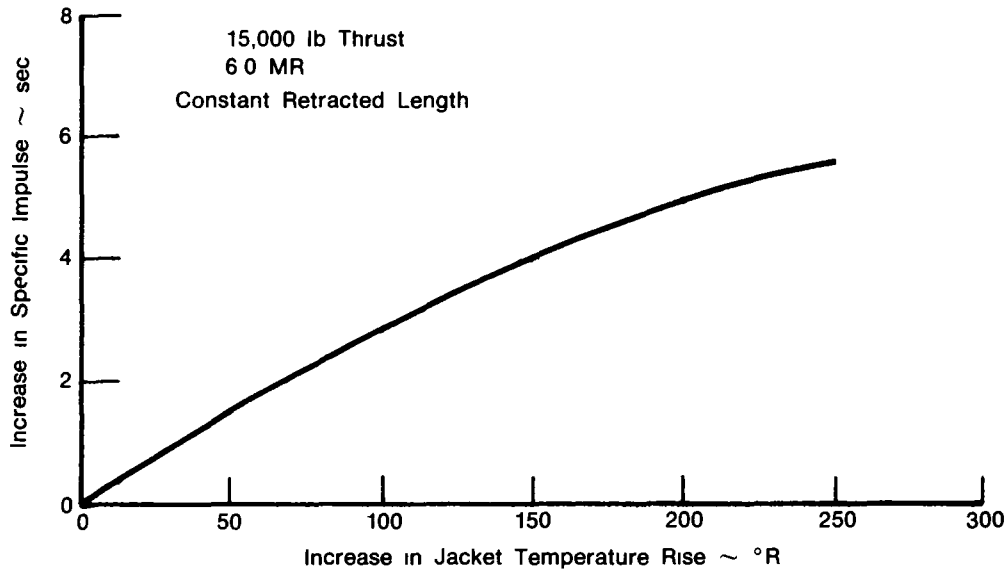
The baseline engine components were analyzed to determine which areas were the most important in trying to improve the engine's capabilities in the direction of the program stated goals which were listed in Section 2.0. This analysis was conducted, in part, utilizing the Pratt & Whitney advanced expander cycle engine optimization program. This computer program provides the capability to reoptimize the engine cycle as individual component and/or cycle characteristics are changed. In this specific analysis, various components characteristics and limits were changed. In regard to the study goals, the results showed that the most critical engine components were the thrust chamber, the turbomachinery, and the extendable nozzle (see Table 3-3).

TABLE 3-3 — BASELINE ENGINE CRITICAL COMPONENTS

|                    |   |
|--------------------|---|
| Thrust chamber —   | Heat transfer, cyclic life, throttle range                                    |
| Turbomachinery —   | Dynamics (critical speed), life/wear (gears, bearings, seals), throttle range |
| Nozzle extension — | Weight, performance (very high area ratio)                                    |

In an expander cycle engine, turbine power is a function of turbine working fluid ( $H_2$ ) temperature. If additional energy could be transferred into the hydrogen, an increase in engine specific impulse would result as shown in Figure 3-10. Additional energy transfer implies a higher thermal conductivity and/or hot wall temperature of the thrust chamber material or an

advanced technology wall geometry (e.g., hot-side fins) Since long engine life is also a technology goal and since higher temperature levels lower thrust chamber low cycle fatigue (LCF) life, the thrust chamber is a critical component of the advanced engine In addition, the thrust chamber must be designed for adequate cooling margin over the entire range of the engine's operation which may also be difficult at certain low thrust levels



FD 254291

Figure 3-10 — Effect of Jacket Temperature Rise on Advanced Expander Performance

The designs of the fuel/turbopump (two stages, 147,000 rpm and 3700 psia discharge pressure) were evaluated to determine if design changes could produce significant engine performance improvements. The most significant change evaluated was that of increasing design point speed. This was done first by holding a constant pump discharge pressure and then by optimizing discharge pressure for main engine performance. Thus, Figure 3-11 shows that approximately 2.5 sec of specific impulse gain could be realized with a doubling of fuel pump speed. However, this speed change will result in significant technology challenges. In particular, turbopump rotor dynamics during both steady-state and transient operation will be extremely critical. Bearing life may also be difficult to achieve in the context of the long life goal of the advanced OTV engine.

In the advanced engine oxidizer pump, increased component efficiencies provide only minimal engine performance gains. The seal between the oxidizer shaft gear which is hydrogen cooled and the oxidizer pump is a life-critical area because a seal failure could result in a catastrophic engine failure. The gear system of the engine is also a life-critical component because of its high pitch-line velocities. Throttle range of the turbopump system is an area of concern since both pumps must provide stable operation over a wide range of engine operating points.

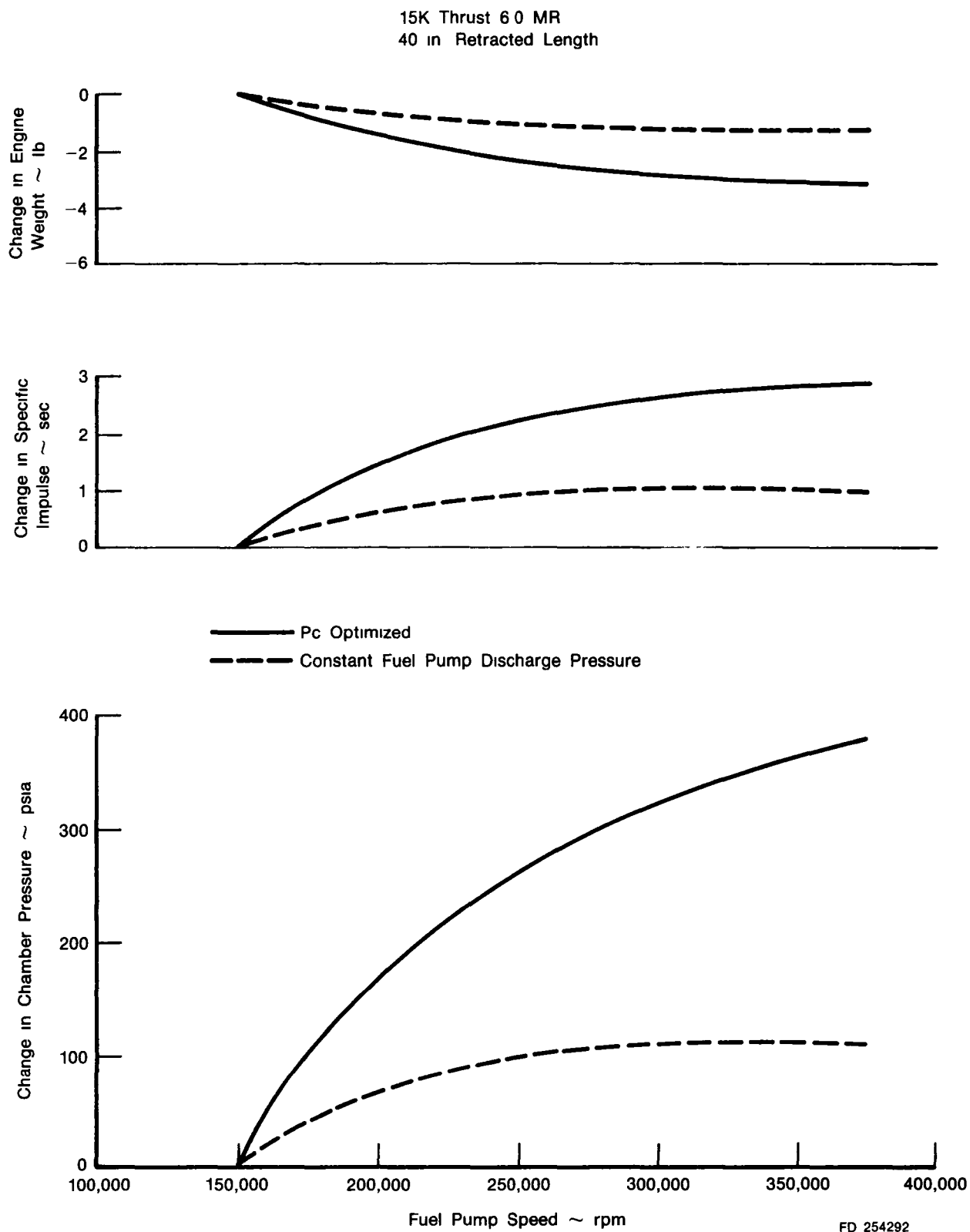


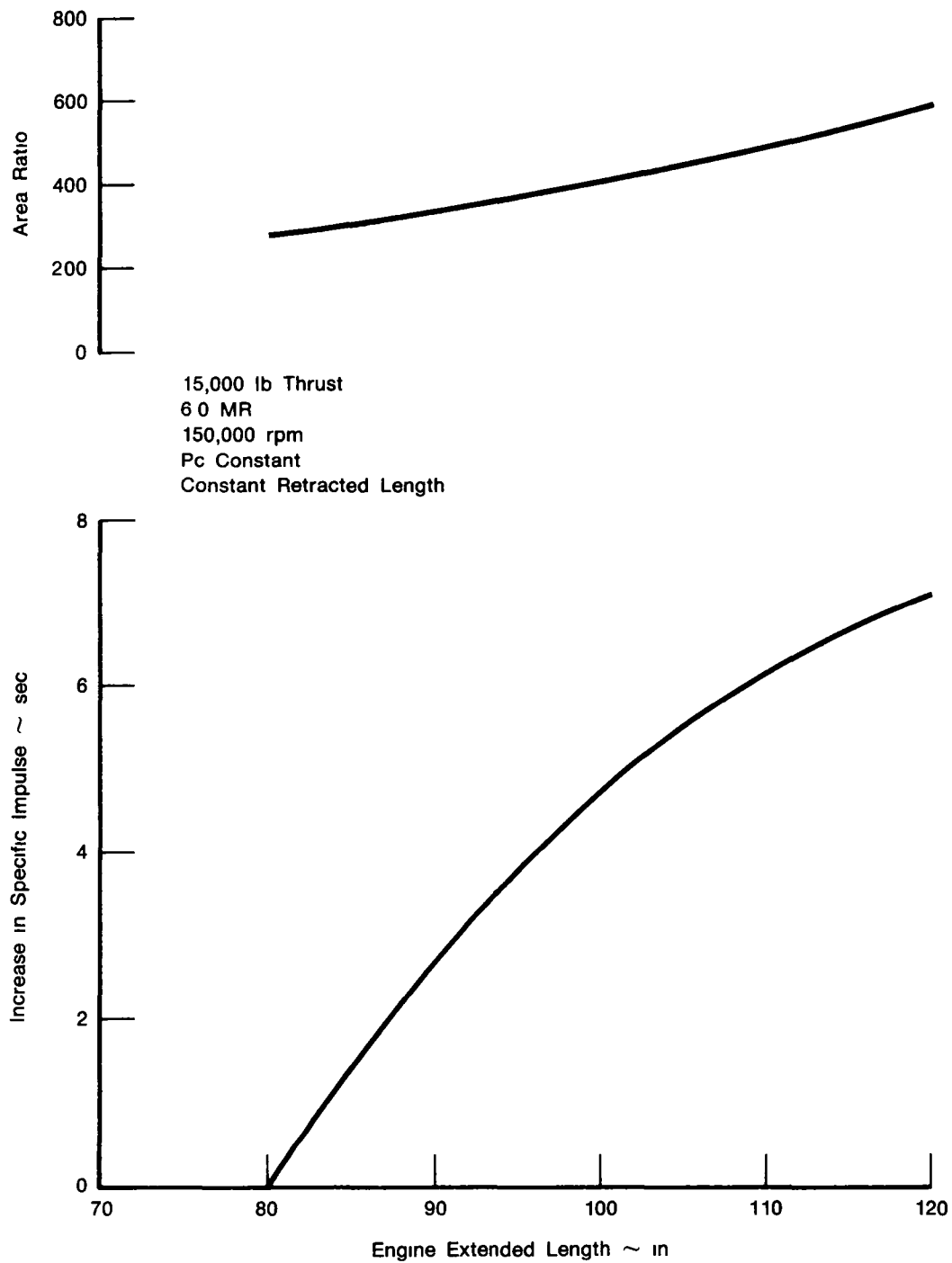
Figure 3-11 — Effect of Fuel Turbopump Speed on Engine Characteristics

A high area-ratio nozzle provides a significant performance advantage for an OTV engine. Typically, such engines have either a fixed nozzle or a single extension section. Since the goal for this study was a 40 in. installed length (rather than the 60 in. of the baseline engine), the conventional single extension would only produce an 80 in. long engine with the nozzle extended. If the engine has a chamber pressure of approximately 1500 psia, the nozzle area ratio would be limited to approximately 280:1. However, the use of multiple section extensions could realize several seconds additional specific impulse as illustrated in Figure 3-12.

The weight of the nozzle extension(s) and the extension mechanism may also be a significant part of the total OTV engine weight. There is also some question of the accuracy of the prediction of specific impulse at very high area ratios since measured data do not exist.

### **3.4 BASELINE ENGINE TECHNOLOGY**

The baseline advanced expander cycle engine was a 1980 state-of-the-art (SOA) design. The technology of a number of its components, such as the turbopump bearings, seals, and gears, the thrust chamber (especially its materials) and overall engine performance, was essentially stretched to the SOA limit. To proceed with confidence into a full-scale development program, these component areas were recommended to be addressed in a technology program in the final report of the prior study (see FR-14615 Advanced Expander Cycle Engine Point Design Study).



FD 254293

Figure 3-12 — Effect of Length on Engine Performance

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## **SECTION 4.0 TURBOMACHINERY TECHNOLOGY**

### **4.1 INTRODUCTION**

Liquid fueled rocket engines use high speed turbopumps to provide the high pressures required at the combustion chamber. The desire to improve pumping efficiency by increasing rotor speed results in an increasingly severe environment for the gears, rotor support bearings, seals, and shafts.

Preliminary designs of advanced OTV engines indicate that drive system components must operate at load conditions and/or rotational speeds beyond the levels of current experience. To meet these requirements, as well as meeting expected operating life requirements, it will be necessary to improve component durability. Gears, bearings, seals, and shafts have the potential of extending the operational limits and life goals.

### **4.2 GEARS**

The conceptual design of the turbopump system of the advanced expander cycle OTV engine (defined under Contract NAS8-33567) is shown in Figure 4-1. This turbopump used a gear system to provide the drive mechanism for the low speed pumps from the oxidizer main pump shaft and utilized a synchronizing gear between the fuel and oxidizer main pump shafts to simplify the control system.

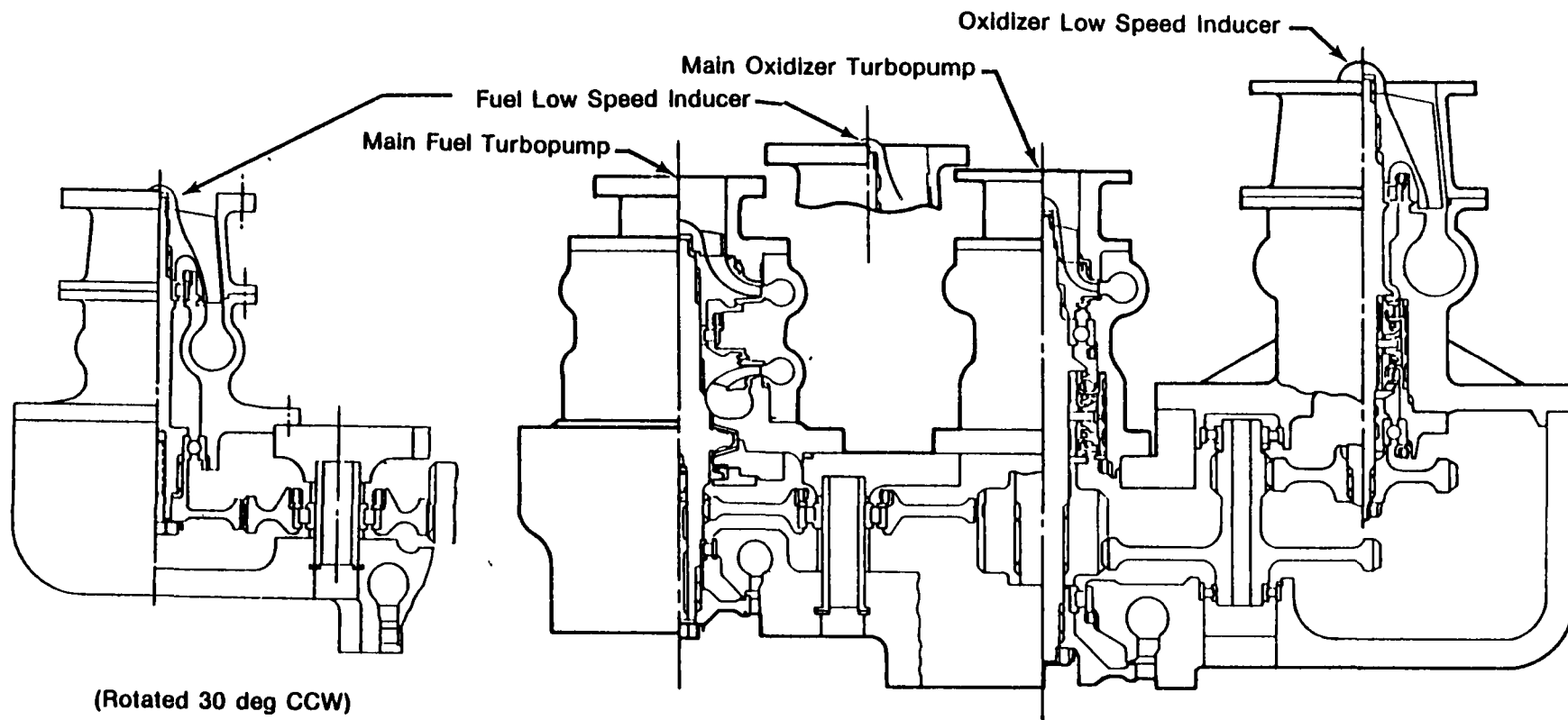
Improved gear durability with resultant longer life for gears operating in a cryogenic environment at higher speeds and loads than on the current engines will provide improved reliability on future cryogenic engines such as the OTV.

The turbopump gear train and its pertinent characteristics are shown in Figure 4-2. This design will serve as a baseline requirement level for the generic characteristics of hydrogen-cooled spur gears.

Under this study effort, a number of potential test candidates which could provide a positive impact on gear durability were identified.

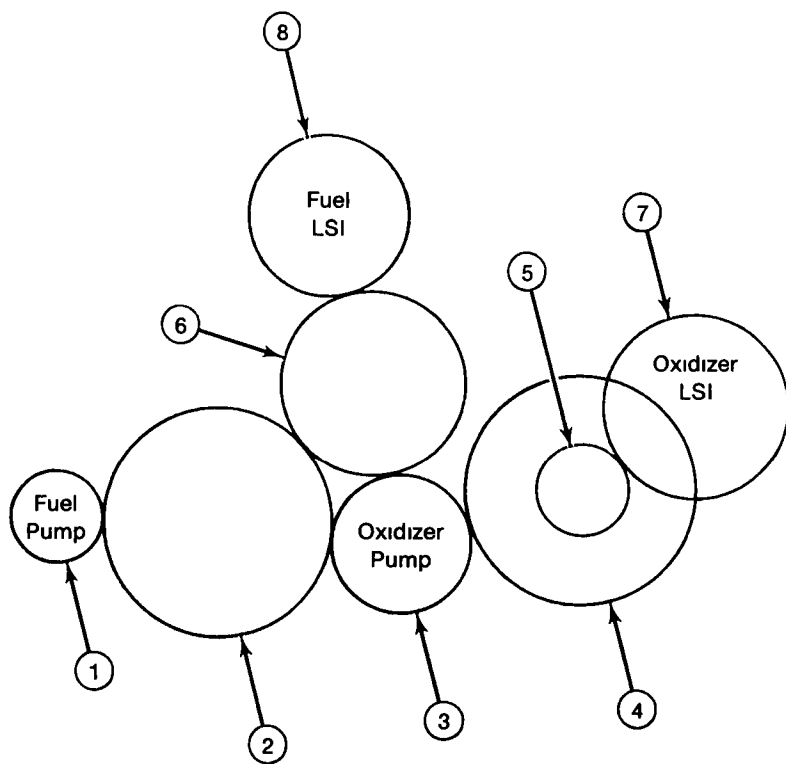
These potential candidates for evaluation fall into five categories:

- Base materials
- Coatings
- Treatments
- Geometries
- Serviceability



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Figure 4-1 — OTV Turbopump Assembly



|                       | Pitch<br>dia<br>(in ) | Pitch Line<br>Velocity<br>(ft/min) | Tangential<br>Load<br>(lb) |
|-----------------------|-----------------------|------------------------------------|----------------------------|
| ① Fuel Pinion         | 1 000                 | 39,300                             | 35                         |
| ② Idler - Fuel/LOX    | 5 000                 | 39,300                             | 35                         |
| ③ Oxidizer Pinion     | 2 230                 | 39,300                             | 35                         |
| ④ Idler - Ox/Ox LSI   | 5 000                 | 39,000                             | 8                          |
| ⑤ Idler - Ox/Ox LSI   | 1 000                 | 7,880                              | 40                         |
| ⑥ Idler - Ox/Fuel LSI | 3 500                 | 39,300                             | 23 5                       |
| ⑦ Ox LSI Pinion       | 3 400                 | 7,880                              | 40                         |
| ⑧ Fuel LSI Pinion     | 3 280                 | 39,300                             | 23 5                       |

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Figure 4-2 — OTV Engine Turbopump Geartrain Characteristics

#### 4.2.1 Base Gear Materials

In addition to establishing a baseline with the current RL10 gear material (AMS 6260), several materials are candidates for a gear durability program

- Standard AMS 6260

Baseline material

- AMS 6265 (premium quality AMS 6260)

Same as AMS 6260 except premium quality, vacuum consumable electrically melted. Less inclusions, etc. Gives potentially better rolling fatigue resistance

- Higher nickel steel (possibly 5 percent versus the 3¼ percent in AMS 6260)
- Rapid solidification rate (RSR) versions of AMS 6260
  - Basic material composition used on RL10. RSR would provide very clean material with no oxides
  - Custom designed powder with possibly lead incorporated for lubricity

#### 4.2.2 Coatings

Several gear coating materials which promise potential life improvement have been identified. The coating materials are as follows:

- Standard MoS<sub>2</sub> with varnish binder

Baseline material

- Variations on the MoS<sub>2</sub> baseline by addition of various elements such as SbO<sub>2</sub> for improved lubricity
- Variations on the MoS<sub>2</sub> baseline by the addition of a PTFE (Polytetrafluoroethylene) powder
- MoS<sub>2</sub> permanently bonded in electroplated layers of nickel-silver (for example, HI-T-LUBE). HI-T-LUBE is a proprietary dry film lubricant (MoS<sub>2</sub>) permanently bonded in electroplated layers of nickel/silver or other compounds with inherent lubricity. Bendix has run several gears at high hertz stresses at -350°F with excellent results and should be good for -400°F hydrogen use
- Electroless nickel (or phosphorous) boron

Plating has excellent hardness and wear properties and can be plated very thin with close tolerances

The coating materials evaluated should consider the method of application and thickness relative to producibility in a development/operational engine

### 4.2.3 Treatments

Two relatively advanced treatment processes (ion implantation and various nitriding processes) appear to show some promise in gear applications

- Ion Implantation

In this process ions of some species are implanted by brute force of acceleration, and these form a near-surface alloy due to dense surface dislocation. The process generally improves wear by a lowered coefficient of friction. The process can be superimposed onto other treatments such as case hardened gears, and at temperatures of 300°F or less. It is a line-of-sight operation, but, no problem is expected.

- Nitriding

For alloyed ferrous materials a nitriding process can obtain a surface which produces a low coefficient of friction, approximately one-third lower than normally expected. The surface has excellent wear properties due to its lubricity, which is not a function of hardness.

### 4.2.4 Gear System Geometries

Basic gear tooth design as well as gear system configuration should be evaluated. For example

- Spur versus helical designs should be considered in which the additional load-causing surface area of the helical gear should be evaluated against its additional induced axial load.
- Tooth crown shaping should be evaluated for potential reduction of gear tooth end loading.
- Tooth contact ratio should be evaluated (changes from the nominal 1.5 to greater than 2 are possible) to reduce gear pressure angle.
- Tooth mesh on both sides of the idler gear can be optimized for a particular design configuration.

### 4.2.5 Serviceability

Long-term service without maintenance might not be necessary if it were feasible to easily reapply the dry film lubricant to the surface of the gear through an access port. Two possibilities are

- Fluorocarbon spray or brush-on (such as Polytetrafluorethylene (PTFE))
- MoS<sub>2</sub> as an additive to an aerosol

## 4.3 BEARINGS

### 4.3.1 Requirements and Current Status

Rotating machinery requires bearings to locate the shaft and to control friction between moving and stationary parts. Rolling element bearings are widely used because of their low friction and tolerance to varying speed and load. Rocket engine turbopumps, using the pumped fluid as a coolant, impose the additional requirements of compatibility with the fluid, tolerance

to cryogenic temperatures, and corrosion resistance. An additional requirement for high speed machinery, typical of turbopumps, is predictable and usually high bearing springrates for rotor dynamics control.

Currently used successful ball and roller bearing designs for cryogenic applications have resulted from a relatively traditional approach. Rolling contact fatigue life prediction is with the Lundberg-Palmgren based equations modified to include centrifugal effects. In most applications the pumped fluid is an unsuitable lubricant but an excellent heat sink. Thus, bearing designs evolved with cages fabricated from materials which form lubricating transfer films. The development of functional designs has depended heavily on empirical guidelines obtained from past experience with both cryogenic and oil lubricated bearings.

A major difference between turbopump bearing applications and conventional applications is the absence of a traditional lubricant. This has been addressed with transfer films formed by sacrificial wear from suitable cage materials. Thus, the mechanisms of transfer film formation and retention and cage wear rate are vital but poorly understood.

The cryogenic temperatures require careful attention to maintain suitable clearances at the rolling contact surfaces and between the cage and its guide surfaces. This in turn produces the need to control both coolant flow and bearing heat generation.

#### **4.3.2 Increased Capability or Reliability**

The desire for improved performance leads to increased rotor speed resulting in a more severe bearing operating environment. Increased rotor speed causes higher centrifugal loads at the element to outer race contact and higher hoop tensile stress in the rotating inner race. In addition, increased durability is required because of the increased number of stress cycles per minute of operation.

This more severe environment combined with the requirement for additional durability makes it imperative that actual environment be well understood. Turbopump rotor support loads are complex and result from a variety of effects. Rotor unbalance response and pumped fluid hydraulic loads can be substantial and are difficult to predict. Load transients and momentary thrust unbalance at shutdown, for example, need to be understood and controlled to a level that does not affect engine operating life.

New turbopump designs requiring increased capability or reliability must include more extensive analyses to fully define the bearing load environment. Examples include the interaction of bearing load on springrate, springrate on rotor unbalance response loads, pump configuration effects on hydraulic load, and pump and turbine thrust balance including start and stop transients. Also, bearing ring stress and fit effects will need better analysis including use of finite element modeling techniques.

#### **4.3.3 Failure Modes**

Rolling element bearings are subject to a variety of failure modes. The traditional design approach has principally addressed rolling contact fatigue because it has been quantified, at least for conventional lubrication applications. This has left other failure modes to be defined by test, resulting in the development of empirical guidelines.

Rolling element bearings are subject to a variety of potential failure modes. These can be grouped into general categories:

- 1 Contact failures which include all types of failure occurring at the element/raceway contact
- 2 Bulk failures of the rings or elements
- 3 Cage failures
- 4 Thermal instability

Contact failures can be surface or subsurface initiated fatigue spalling or element and raceway wear. Subsurface initiated fatigue is the classical failure mode expected for all rolling element bearings. Because of the historical attention given to subsurface fatigue, there are well established life prediction techniques for oil lubricated bearings with the result that subsurface fatigue failures are rare. Cryogenic bearings are assumed to behave in a similar manner but extra fatigue life margin is usually attributed to account for the unknown fatigue properties of the standard 440C material at cryogenic temperatures. Surface initiated fatigue may result from handling, contamination damage, or surface distress. Properly operated rolling contact bearings require a lubricant film to separate the highly stressed contact surfaces. Lack of film separation will result in surface damage leading to gross wear or spalling. Cryogenic bearings depend on the formation of a solid film transferred to the elements and raceways by sacrificial wear from the cage. These films are sensitive to surface motion and stress at the bearing raceways and may be incapable of sustaining a lubricating function if their operating limitations are exceeded. Abrasive contamination will cause wear at the contacts and some cage fillers may abrade the rolling elements.

Bulk failures may be either plastic deformation or fracture. Severe overload, even momentary, will result in plastic deformation and failure with continued operation. Overload damage will be most severe when the elastic and plastic deformations force the contact areas beyond the race extremities, a thrust overload that forces the ball contact ellipse above the race shoulder for example. Currently used bearing steels have low fracture toughness with the results that bulk stress, well below the material capability, can cause fracture in the presence of small defects. Ring bulk stress results from centrifugal force due to rotation and ring-to-shaft interference fit pressure. Defects with the potential for initiating fracture can be caused by rolling contact fatigue spalls, high cycle fatigue cracks, mounting surface fretting or damage, and manufacturing defects.

Rolling element bearing cages separate the rolling elements to avoid the high element to element rubbing that would otherwise occur. Additionally, in cryogenic bearings the cage material provides bearing lubrication by sacrificial wear. Cage loads at the element pockets and the guide lands must be controlled to prevent fracture or excessive wear. Operating speed, external load magnitude and direction and bearing design all contribute to the cage loads. Cryogenic bearing cages require structural support either with a metal shroud or fiberglass fabric within the matrix.

Bearing operating internal clearance must be maintained over a relatively narrow range for proper operation. Loss of clearance generates the high internal load and can cause seizure. Temperature of the bearing rings and mating parts controls bearing clearance, thus, thermal stability must be maintained. Rolling element bearings are low friction devices, nevertheless, they are significant sources of heat. The cooling system must have the capability to maintain stable temperatures during operation. Inadequate cooling or momentary loss of cooling can lead

to loss of clearance with a consequent increase in heat generation. This form of thermal instability will cause rapid failure.

Increasing speed results in higher element centrifugal loads and more stress cycles with time. In addition, rotor unbalance and rotor dynamic response generally contribute more load with increasing speed. Thus, subsurface fatigue will become more likely. This will require a better understanding of the rolling contact fatigue properties of current materials and may require new materials with improved life.

Cryogenic bearings depend on a transfer film between the contacting surfaces to prevent metal-to-metal contact and surface distress. The higher stress combined with the higher sliding velocity at the contact will adversely affect this film. The increased difficulty in maintaining film separation may result in premature failure due to microspalling or smearing at the contacting surfaces progressing to severe material removal by wear or gross spalling.

The fracture sensitivity of currently used bearing steels may prohibit significant increases in rotor speed. Experience with current materials has repeatedly shown that ring fracture can occur when ring tensile or hoop stresses exceed about 30,000 psi. Speed induced centrifugal stresses are unavoidable and alternate materials with improved fracture toughness or novel mounting arrangements to compressively preload the rotating ring will be required.

Bearing cages will also be more severely loaded with increased rotor speeds. The classical failure mechanisms of guide land wear, pocket wear, side or cross rail fracture and delamination of composite structures will be more likely to occur. Increased bearing speed increases the sliding velocity and cage centrifugal load and has the potential for greatly increasing the element to cage and cage to guide land loads. These higher loads will require stronger cage materials or composite construction with an emphasis on structural integrity. The higher sliding velocity, and higher loads, also cause an increase in wear rate. These materials must wear to form the required transfer lubricant film but wear rate must be controlled so that cage integrity is maintained for the desired bearing life.

Increased bearing speed results in higher heat generation and at the same time, the increased windage and internal churning reduces cooling effectiveness. Thus, achieving thermal stability will become more difficult. Design modifications to reduce heat generation may be required. Early ball bearing designs used open race curvatures on the inner race which minimized ball-to-race spin heat generation at the expense of increased hertz stress. In addition, more effective cooling techniques to assure coolant flow at the point of heat generation will be required. Cryogenic fluids are excellent coolants but the volume changes associated with temperature can locally impede flow in areas such as the ball/race contact.

#### **4.3.4 Technology Shortfalls and Recommended Investigations**

Advanced rocket engine turbopumps will require improved bearings to reliably achieve improved performance with increased life. Improvements in the areas of rolling contact durability, toughness, transfer film lubrication technology, and cooling technology will be required.

Any material evaluation program must begin with testing of 440C to provide a comparative baseline. Preferably this would include back-to-back comparison of conventional oil lubricated bearings and cryogenic bearings. Initial fatigue screening tests could be with element testers using either ball or rod specimens but final material selections should be based on full-scale bearing tests.

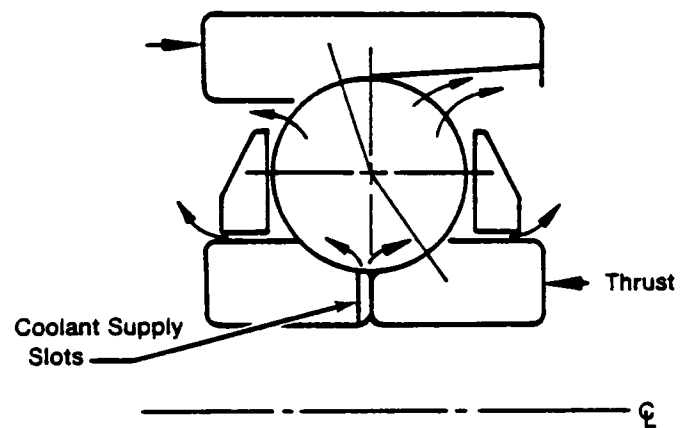
Candidate materials are available from three categories: conventional through-hardened stainless steels, carburizing alloys, and ceramics. Conventional alloys with the potential for improved durability include powder processed (especially rapid solidification rate — RSR) 440C, CRB7, and BG42. These alloys, or processing variations, have shown improved rolling contact fatigue performance but little or no improvement in toughness. Carburizing alloys are surface hardened only, leaving a tough ductile core which improves fracture resistance. Potential carburizing alloys include 9310, CBS600, CBS1000M, and an experimental low carbon variation of BG42. All but BG42 are low chrome alloys and would require chrome implantation for corrosion resistance. The only ceramic which has shown potential for use in bearings is silicon nitride ( $\text{Si}_3\text{N}_4$ ). Silicon nitride may be used advantageously for rolling elements because of its low mass (41 percent steel) but fracture sensitivity precludes its use for rotating rings.

Cryogenic bearings rely on self-lubricating cage materials to form transfer films by sacrificial wear. The increased speed and stress of future turbopump bearings will require transfer film technology development. Friction and wear data, as may be generated with a pin on disk type tester, is required for comparison of various candidate materials. A five ball tester, using cages made from candidate materials, is suitable for lubrication evaluation. This type of tester provides the capability for varying speed, stress, and contact slip.

Armalon, a fiberglass fabric reinforced Teflon, is used in current cryogenic applications and should be the initial test specimen to provide a comparative baseline. Other Teflon compounds such as Rulon A and Salox M have been successfully used and are logical candidates. In addition, other fluoroplastics should be surveyed for potential suitability as candidate materials. Also, metals such as lead and silver which have had some success when plated on steel, should be surveyed for potential use either as coatings or fillers in fluoroplastics.

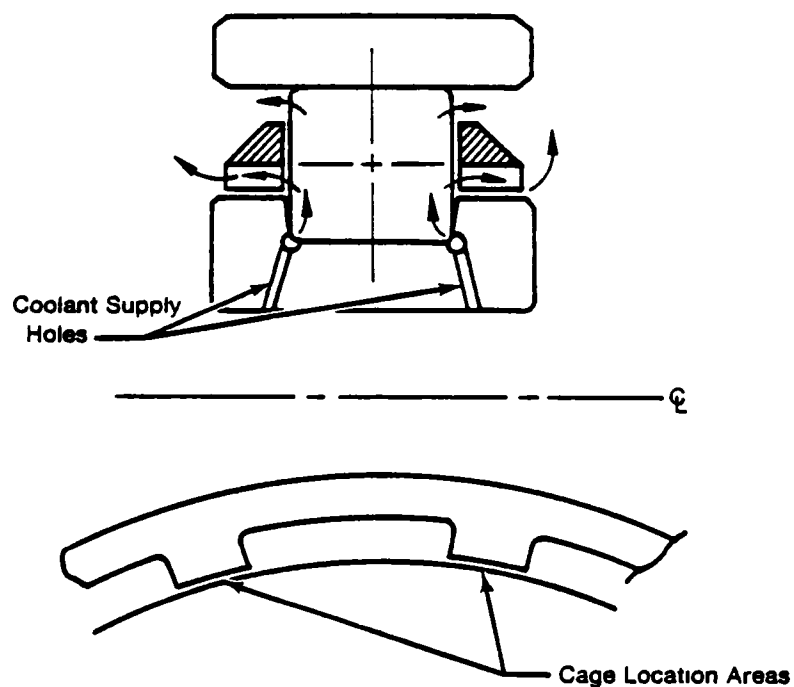
Cooling effectiveness can be increased with designs which decrease flow resistance or direct the coolant into the areas of mechanical contact. Through race cooling has been effectively used with oil lubricated bearings to inject the coolant into the ball/race contact area. This may also be beneficial for cryogenic bearings. Commonly used cage designs in cryogenic bearings have large cross sections for structural integrity, which impedes coolant flow and results in undesirable churning. Use of ball bearings with counter bored races, if the thrust load is unidirectional, or reinforced cage side rails with a reduced cross section, are techniques which can be used to increase flow area.

Potential design variations which address some of these technology requirements are shown here. Figure 4-3 is a split inner race design with coolant entry slots at the split face combined with a counter bored outer race, both providing improved coolant flow. Figure 4-4 shows under race cooling applied to roller bearings combined with a skeletonized cage to increase flow area. Figures 4-5 and 4-6 show a potential design to reduce tensile stress in the fracture sensitive race material by compressively preloading with structural side rails. In addition, Figure 4-7 shows use of a roller guiding cage preventing sliding contact between the roller ends and the guide flanges, a problem area with earlier roller bearing designs.



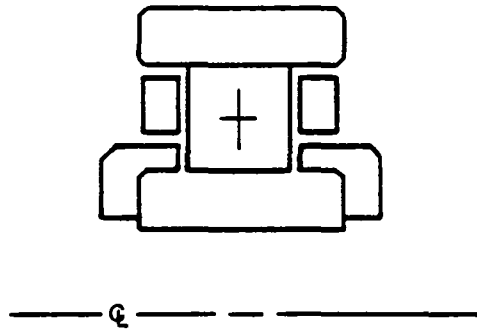
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Figure 4-3 — Angular Contact Bearing With Inner Race Supplied Coolant



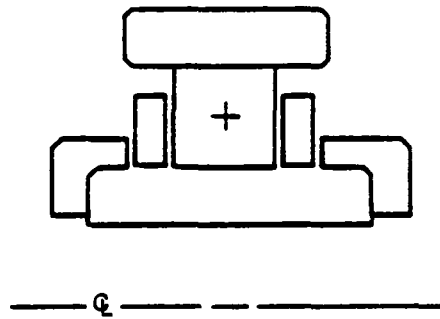
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Figure 4-4 — Cage With Reduced Guide Land Contact



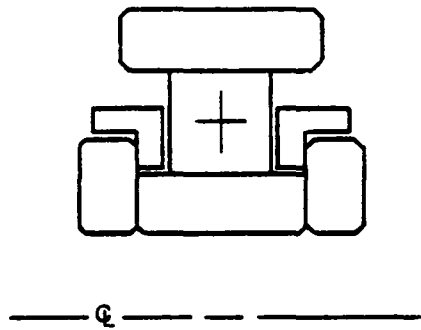
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Figure 4-5 — Prestressed Inner Ring



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Figure 4-6 — Prestressed Inner Ring With Cage Guided Roller



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Figure 4-7 — Cage Guided Roller

## 4.4 SEALS

A very important element in the successful development of high performance turbomachinery is the selection of the proper rotating shaft seals. This is especially true of the seal separating oxygen at the back of the lox pump impeller from the hydrogen in the gearbox. A program related to shaft seal technology would fall into the following areas: material properties, seal configuration, and dimensional control and/or design operation. These areas are discussed in the following paragraphs.

### 4.4.1 Materials

- a Nosepiece/ring seal — most current candidates are carbon based and include the following

- P5AG baseline

This silver impregnated carbon is presently used in all RL10 seals and is relatively expensive. It is commonly used in high-bearing load applications.

- P5N (has a film former)

This carbon seal has been used in several rocket applications.

- P692 (has a resin filler)

- CDJ-83 (high temperature treatment)

This material has been used in several rocket seal applications.

- CJPS (high temperature treatment)

This material is made from nonexuding graphitized base stock and features a reduced running coefficient of friction and better oxidation resistance.

- Graphitar 3048 (high temperature resin treated)

This graphitized carbon is used in the PWA 2037 gas turbine engine. It is nonexuding and has improved sliding lubricity and good oxidation resistance.

- Graphitar 84 (resin treated)

- SP-3 and SP-11

Two polyimide plastics that are considered viable candidate materials.

- b Seal plates

- Chrome plated steel (currently used on RL10)

- Electroless nickel plated steel
- Electroless nickel boron plated steel
- Chrome carbide plated steel
- Nickel alloy steel with surface infused PTFE

This is comprised of a hard nickel alloy plating with the surface subsequently infused with PTFE fluorocarbon

- MoS<sub>2</sub> family coated steel

Many MoS<sub>2</sub> base lubricants are available as candidates to provide potential improved seal plate durability

The plating/coating materials evaluated should consider method of application and thickness relative to producibility in a development/operational program

#### 4.4.2 Seal Configuration

There are a large number of different potential seal types which warrant consideration as candidates for testing in a technology program (Some 15 types are given in the NASA Monograph "Liquid Rocket Engine Turbopump Rotating Shaft Seals," NASA SP-8121.) These cover the range from modifications to existing face seals to new seal designs much more complicated from a mechanical standpoint than the current face seals. New seal types generally fall into the category of face or circumferential and hydrodynamic or hydrostatic seals or hybrids thereof.

Two recent reports, "High-Speed Cryogenic Self-Acting Shaft Seals for Liquid Rocket Turbopumps," NASA CR-168194, June 1983, and "Small High-Speed Self-Acting Shaft Seals for Liquid Rocket Engines," NASA CR-135167, September 1977, discuss recent high-speed cryogenic advanced seal technology experience. The later report (NASA CR-168194) covered the design analysis, fabrication, testing, and evaluation of five potential self-acting lift pad seals. Three of these seals resulted in a favorable test life with low leakage rates and should be considered in the design of advanced OTV type engines.

Even with technical problems of these and other seals, a recommendation of potential seal configurations would be the result of a more detailed study of the OTV gearbox operating conditions and environment combined with applicable seal technology experience.

#### 4.4.3 Dimension Control/Design Operation

For the best possible understanding of seal performance (wear and leakage), it is necessary to understand not only the effects of dimensions but also distortions and pressure balancing.

- *Surface Finish* — Usually flat within 0.00002 with a 5 or 10 finish. Review requirements and seal vendor recommendations.
- *Squareness* — Usually square within 0.0005 with the ID. An in-depth study of squareness of the mating surface and seal face is required.
- *Radial Location (Concentricity)* — As with squareness, a complete study should be made.
- *Misalignment* — Seal must compensate for inertia effects due to misalignment. There are no available guidelines.

- *Stationary Seal or Nosepiece Distortion* — Determine the effects of thermals on the sealing face. Innovative design might result in compensations in the seal assembly design which minimize seal nosepiece distortions during cryogenic operation
- *Seal Plate Distortions* — Determine the effects of thermals, assembly, and dynamics on the sealing plate face. As on other engines, proper design can hold distortions to an absolute minimum. Under this category, effects of seal/plate vibrations should be considered
- *Pressure Balancing* — Determine pressure loads versus wear over the operating range

#### 4.5 RAPID SOLIDIFICATION RATE TURBOPUMP SHAFT

Advanced OTV turbopump designs require very high rotational speeds in order to achieve high efficiency levels. This high-speed requirement in turn produces an increased concern over shaft critical speed margins and bearing loads. These concerns could be somewhat reduced (or higher speeds could be run with existing margins) if the material properties of the main shaft could be improved.

The two potential benefits of a stiffer turbopump shaft made with the RSR process are related to the concern about critical speed. If the reduced concern were traded for an increased rotational speed, higher pumping efficiencies and, therefore, increased engine performance could result. The P&WA/GPD Advanced Expander Optimization Program was run to determine the effect on performance with increasing fuel turbopump speed without regard to material, stress, and other similar limitations. The result of these cycle cases shows that if system pressures are held constant, only about one second of specific impulse (Isp) gain may be realized. Further, if system pressures are not limited, an additional two seconds of Isp could be achieved. However, very large increases in rotational speed are required and the attendant stresses, bearing loads, etc. are unlikely to be within 1990 state-of-the-art technical capability.

The second approach to the reduced critical speed concern is to hold turbomachinery speed and pressure levels constant and accept the increased margin as a reduction in program risk. An analysis was conducted to determine the effect on fuel and oxidizer turbopump critical speed margins by the substitution of a stiffer mainshaft material. The previous design analysis (under Contract NAS8-33567) assumed conventional materials for the rotor shafts ( $E = 31 \times 10^6$ ,  $\rho = 0.298 \text{ lb/in}^3$ ). A new, stiffer material ( $E = 38 \times 10^6$ ,  $\rho = 0.308 \text{ lb/in}^3$ ) has been developed which should produce some improvement in critical speed margins if a direct substitution were accomplished. Table 4-1 summarizes the results of the material substitution analysis.

TABLE 4-1 — RESULTS OF MATERIAL SUBSTITUTION ANALYSIS

| Pump     | Design<br>Operating<br>Speed<br>(rpm) | First Critical Margin<br>Over Design Speed<br>(%) |                 | Bearing<br>Load Change<br>at Operating<br>Speed<br>(%) |
|----------|---------------------------------------|---|-----------------|--|
|          |                                       | Original<br>Material                              | New<br>Material |  |
| Fuel     | 150,000                               | 25  | 27              | -10  |
| Oxidizer | 67,400                                | 16  | 17              | -4   |

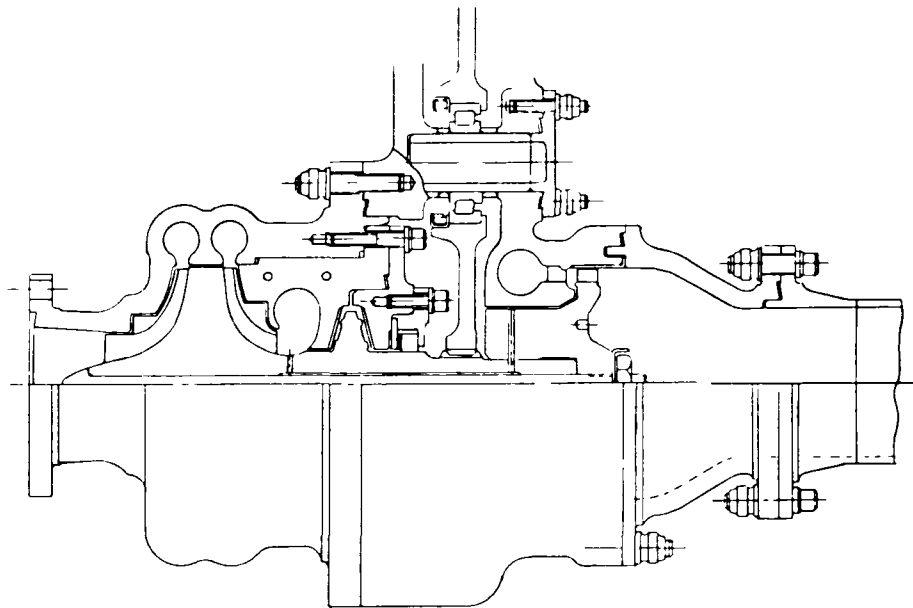
Material technology work currently in progress at P&WA/GPD has produced materials which promise significant improvements over those currently available for turbopump rotor

shafts For example, a 20 percent improvement over current shaft materials in stiffness-to-density ratio has been demonstrated and a potential improvement of 40% has been indicated

#### 4.6 ADVANCED MATERIAL TURBOPUMP TECHNOLOGY PROGRAM

The use of an advanced material for a totally nonmetallic fuel turbopump appears to offer the potential of a high-strength, low coefficient-of-expansion machine If such a concept is feasible, much higher turbopump speeds are possible with a reduced concern in such areas as critical speed, tip stresses, bearing loads, and differential thermal growth In addition, the engine weight can be reduced

A conceptual design of an advanced technology lightweight material turbopump is shown in Figure 4-8 This design will serve as the baseline to determine whether a composite material turbopump is practical The initial effort needs to be directed towards obtaining applicable composite material data from the P&WA Advanced Materials Engineering Group and then supplementing it with various outside sources for more detailed information



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Figure 4-8 — Advanced Technology Material Turbopump

Advances in bearing and seal technology may be required in order to achieve low weight and high pump and turbine efficiencies The use of hydrostatic or hydrodynamic fluid-film bearings and seals must be analyzed, so as to extend turbopump life and performance capability Investigations conducted to date, both in rocket and gas turbine engines, must be reviewed to identify current design limitations

A turbopump layout, summarizing the results of a detail component and material design study, will enable an evaluation of benefits in regard to weight, critical speed, life capability, and performance to be conducted

The effort necessary for the successful advancement of the technology involved in an advanced lightweight turbopump will involve design study trades of the identified benefits (e g ,

rpm versus critical speed, weight versus number of stages, etc ) Then the specific material selections can be made and a fabrication technology effort can be initiated

## SECTION 5.0

### THRUST CHAMBER/NOZZLE TECHNOLOGY

Two key components of the OTV engine system are the thrust chamber and the nozzle with its extension system. Material properties of the thrust chamber are the primary cycle-life limiting engine component. A high area ratio nozzle is required for the high performance (specific impulse) needed by the advanced OTV engine. The P&WA designed thrust chamber (as defined under Contract NAS8-33567-OTV Engine Point Design Study) was based on the available property data of aged AMZIRC, a high thermal conductivity copper-zirconium alloy. This alloy has improved thermal fatigue and strength properties over pure copper but at a small loss in thermal conductivity.

In an expander cycle engine, the cycle power is dependent on the turbine working fluid energy which is equal to thrust chamber/nozzle coolant exit energy. Higher coolant exit temperatures, however, are limited by the thermal conductivity of the wall material and the hot wall temperature of the thrust chamber as well as coolant velocity. Higher wall temperatures also decrease chamber cyclic life.

There is substantial scatter in the available copper-alloy cyclic life property data. Since standard design practice is to use a factor of four times the lower bound of the data, the design configuration will probably be inferior to the allowable optimum utilization of the material.

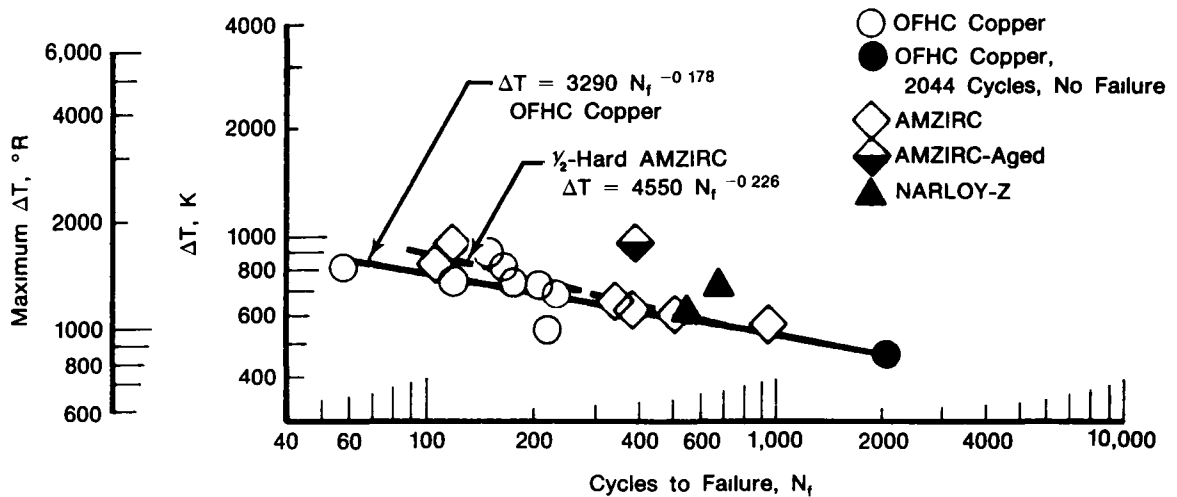
Pratt & Whitney believes that a technology program to design an optimized thrust chamber for an advanced engine should begin by an investigation to obtain and optimize the best possible copper-base alloy for a high heat transfer thrust chamber by improving cyclic life and thermal conductivity material properties.

#### 5.1 THRUST CHAMBER

The LCF lifetime of several copper base alloys has been investigated by both P&WA/GPD\* and NASA\*\*. The basic problem discussed in both P&WA/GPD and NASA references can be related to inadequate cyclic life for AMZIRC in the half-hard strain hardened condition. It is apparent from metallurgical information contained in the NASA reference that AMZIRC is not microstructurally stable during the accumulation of strain and temperature during use. Intense plastic deprivation, induced by thermal cycling, is responsible for the accumulation of damage by intense slip. The following heat cycle apparently recovers the microstructure sufficiently for the next deformation cycle to produce more damage, and the material undergoes cyclic strain softening with the deformation strain per cycle accumulating until failure. This type of behavior is shown in Figure 5-1b and is based on the published stress-strain hardening for AMZIRC at 900°F. An intrinsically stronger alloy with a lower propensity for thermally induced recovery and/or recrystallization during the exposure cycle should exhibit a lower total strain accumulation and therefore superior cyclic life capability. Figure 5-1 shows this to be the situation for aged AMZIRC and the alloy Narloy-Z. Both these alloys should therefore be considered to be improvements over half-hard AMZIRC.

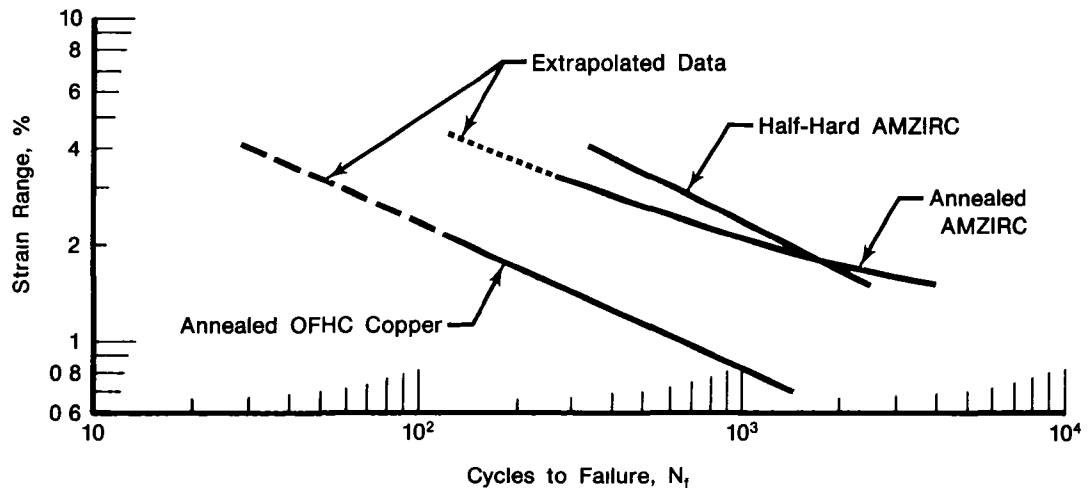
\* Jaeger, R., "Low Cycle Fatigue Analysis of OTV Thrust Chamber," P&WA/GPD Design Notes 80-712-3288, July 1980.

\*\* Quentmeyer, R. J., "Experimental Fatigue Life Investigation of Cylindrical Thrust Chambers," NAS TMX 73665, July 1977.



Maximum Hot-Gas-Side to Backside Wall Temperature Difference,  $\Delta T$  versus Cycles to Failure for the OFHC Copper, AMZIRC, and NARLOY-Z Cylinders

(a)



Typical Low-Cycle Fatigue Life of Annealed OFHC Copper, Annealed AMZIRC, and Half-Hard AMZIRC  $\dot{\epsilon} = 0.002 \text{ sec}^{-1}$ , 538 K (1460 $^{\circ}\text{R}$ )

(b)

FD 254296

Figure 5-1 — LCF Data for Several Copper-Base Alloys

Detailed alloy design for this application principally requires an elevated temperature alloy approach, where more strain can be accommodated elastically than the normal AMZIRC capability. It is possible that several alloy systems could produce such elevated temperature capability.

### 5.1.1 Copper — Zirconium

Rapid solidification rate processing offers the prospect of increasing the level of zirconium in the alloy and improving the degree of alloy homogeneity. The maximum level of zirconia currently available is 1.5 percent (AMZIRC = 0.15 percent), using RSR the zirconium level could possibly be raised to levels of 5 to 7 percent. The resulting precipitation hardened alloy, using  $\text{Cu}_3\text{Zr}$  rather than zirconia, would exhibit higher yield and fatigue strength, higher thermal conductivity and a lower density.

### 5.1.2 Martensitic — Copper — Aluminum

There are a wide range of aluminum bronze alloys commercially available, for which strengthening by the  $\beta$  martensitic reaction is possible. These alloys show both high strength and toughness and alloy modifications by RSR would allow some of the conventional alloying constraints to be removed. The alloy family to be investigated is the high temperature bronze group ( $\text{Cu-10Al-5Ni-2.5Fe-X}$ ) with grain boundary stabilization by intentional dispersed phases such as  $\text{TiN}$ ,  $\text{TiB}_2$ , etc.

### 5.1.3 Copper — Chromium

Copper alloys containing chromium are generally limited to the eutectic composition, 1.28 wt % Cr. Rapid solidification could extend the Cr content up to 10 percent, producing alloys which are analogous to the elevated temperature aluminum alloys in high temperature performance. These alloys will show elevated temperature strength and fatigue performance, but require some detailed optimization for full property potential.

### 5.1.4 Copper — Zirconium — Silver

These alloys (Narloy-Z type) are currently used in industry and offer the same prospect for improvement as discussed under Item 5.1.1 above.

The problems relating to copper base alloy production, and the inherent performance restrictions which are imposed on thrust chamber design by the use of conventional alloys are well known. An alternate approach which would overcome these problems relies on the use of rapid solidification process equipment for which major areas of applicability to such an investigation are

- Use of inert helium gas processing, thereby limiting the opportunity for copper base alloys to be exposed to oxidation and hydration reactions
- Use of rapid solidification inherent in this process to produce substantial supersaturation of ternary or quaternary element additions, with the objective of producing the necessary secondary dispersed phase distributions for planar-slip inhibition and recrystallization control.

Pratt & Whitney Aircraft has conducted program feasibility studies by producing copper base alloys for both Naval Research Laboratories and Air Force Wright Aeronautical Laboratories (AFWAL). These alloys were substantially cleaner and more ductile than alloys of comparable composition from other sources.

## 5.2 NOZZLE EXTENSION SYSTEM

High area-ratio nozzles provide a significant performance advantage for upper-stage rocket engines. Typically, such engines have either a fixed nozzle or a single extension section. The use

of multiple section extensions, however, could realize several additional seconds of specific impulse for length constrained OTV application

The goal/baseline engine for this study has a 40 in installed length. In order for the OTV to be carried in the space shuttle bay with a minimal impact on vehicle length, it is desirable for the OTV engine to be shortened from 120 in to 40 in. A conventional single extension would only produce an 80 in long engine with the nozzle extended. A 40 in installed length engine with a 120 in extended length can be accomplished by segmenting the nozzle and retracting the segments forward around the engine components. The smaller segment requires only a conventional linear actuator but the two larger segments required more stroke than can conveniently be stored in a conventional actuator. A literature review revealed an actuator which demonstrated a large extension from the stowed length and was applicable to retractable rocket nozzles.

The upper nozzle segment is regeneratively cooled in order to provide sufficient hydrogen coolant energy pick-up to provide a sufficiently high chamber pressure ( $\sim 1500$  psia) in order to hold the overall engine geometry.

The minor diameter (front diameter) of the middle retractable nozzle segment must pass over all engine components in order to stow in the required position. This requirement necessitated relocating many of the components and rerouting the plumbing.

The geometry of the actuators and linkage may produce an interference between the actuator and front end of the nozzle. Since this effort was a conceptual design, the detailed geometry was not optimized.

The actuator cylinders are double walled in order to route the actuating gas to the retraction side of the piston which is the outer annulus of each cylinder. The extension side is the inner cavity (Figure 5-2).

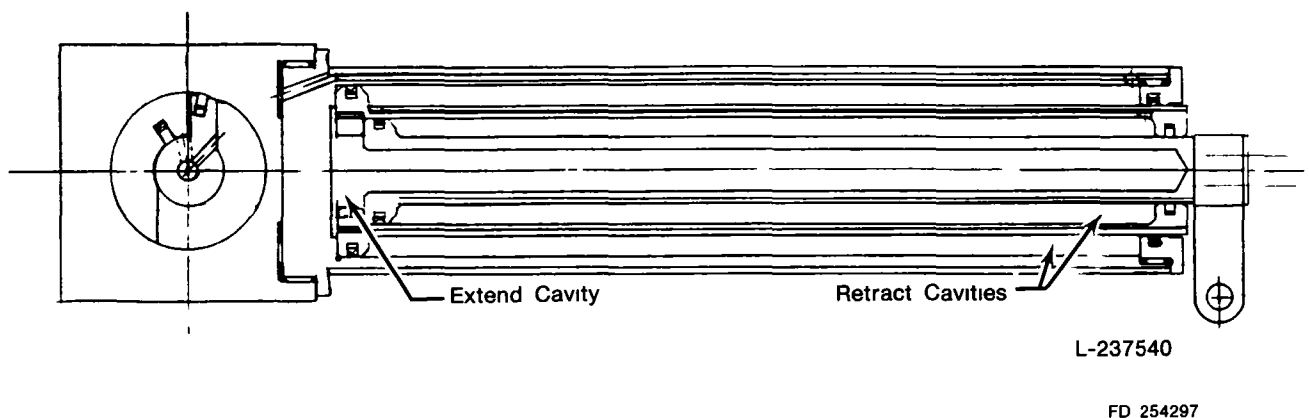
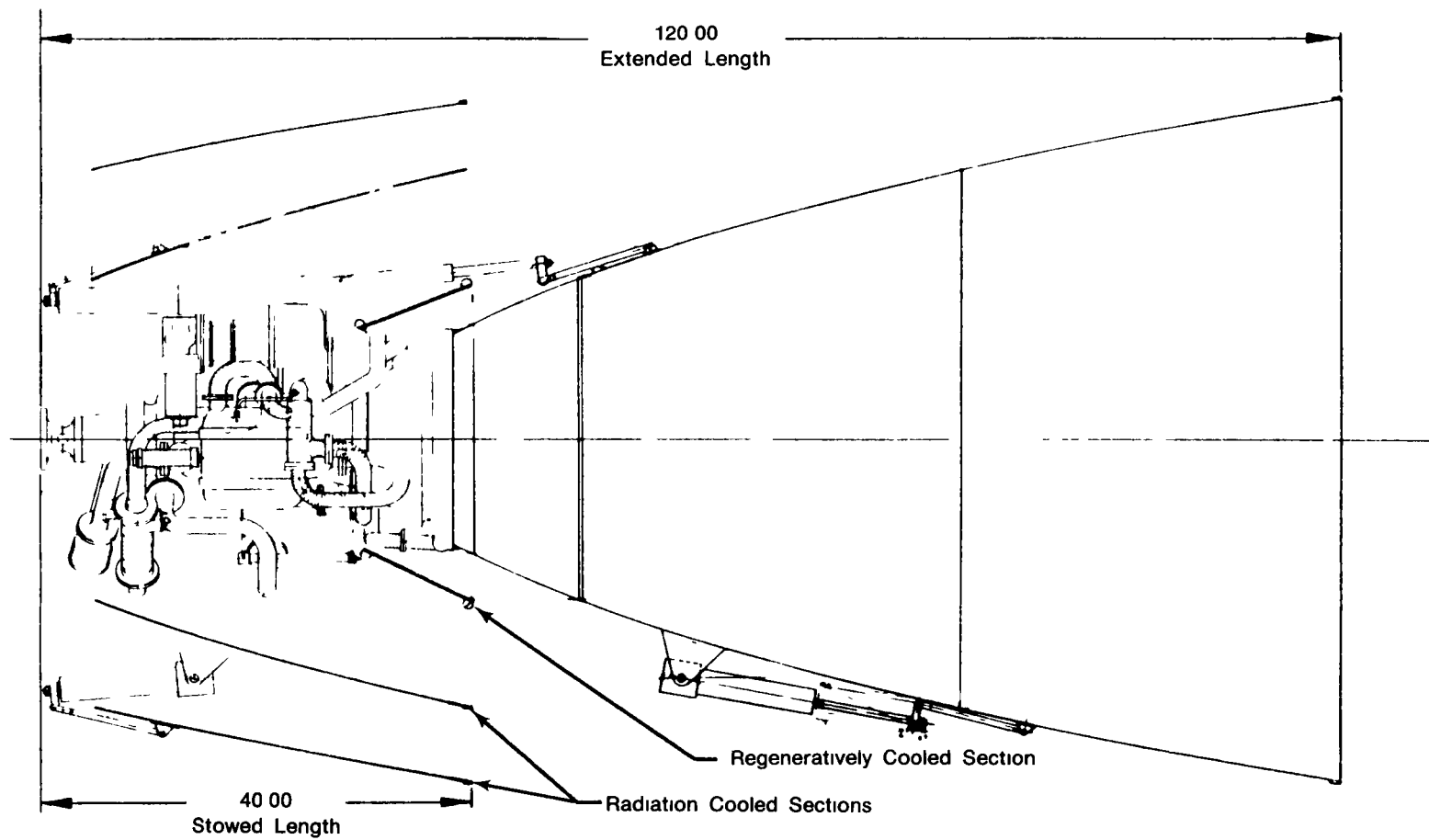


Figure 5-2 — Rotary-Linear Actuator

The uncooled nozzle segments were considered to be carbon/carbon. The conceptual design of a lightweight multisection nozzle and its extension mechanism is shown in Figure 5-3.



FD 253280A

Figure 5-3 — Advanced Nozzle Extension System

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## SECTION 6.0

### CONTINUOUS THROTTLING CONCEPTS

Recent studies have shown a benefit for an OTV delivery of a large space structure from low earth orbit (LEO) to geosynchronous earth orbit (GEO) if the OTV could operate at a constant acceleration level rather than a fixed, low thrust level. (See NASA CR-168277, "Primary Propulsion/Large Space Systems Interactions Study," Martin-Marietta Denver Aerospace, December 1981 ) This benefit could only be realized if the OTV engine could be continuously throttled, down to a level of approximately 500 lb (up to 30:1 relative to a full thrust level of 15,000 lb)

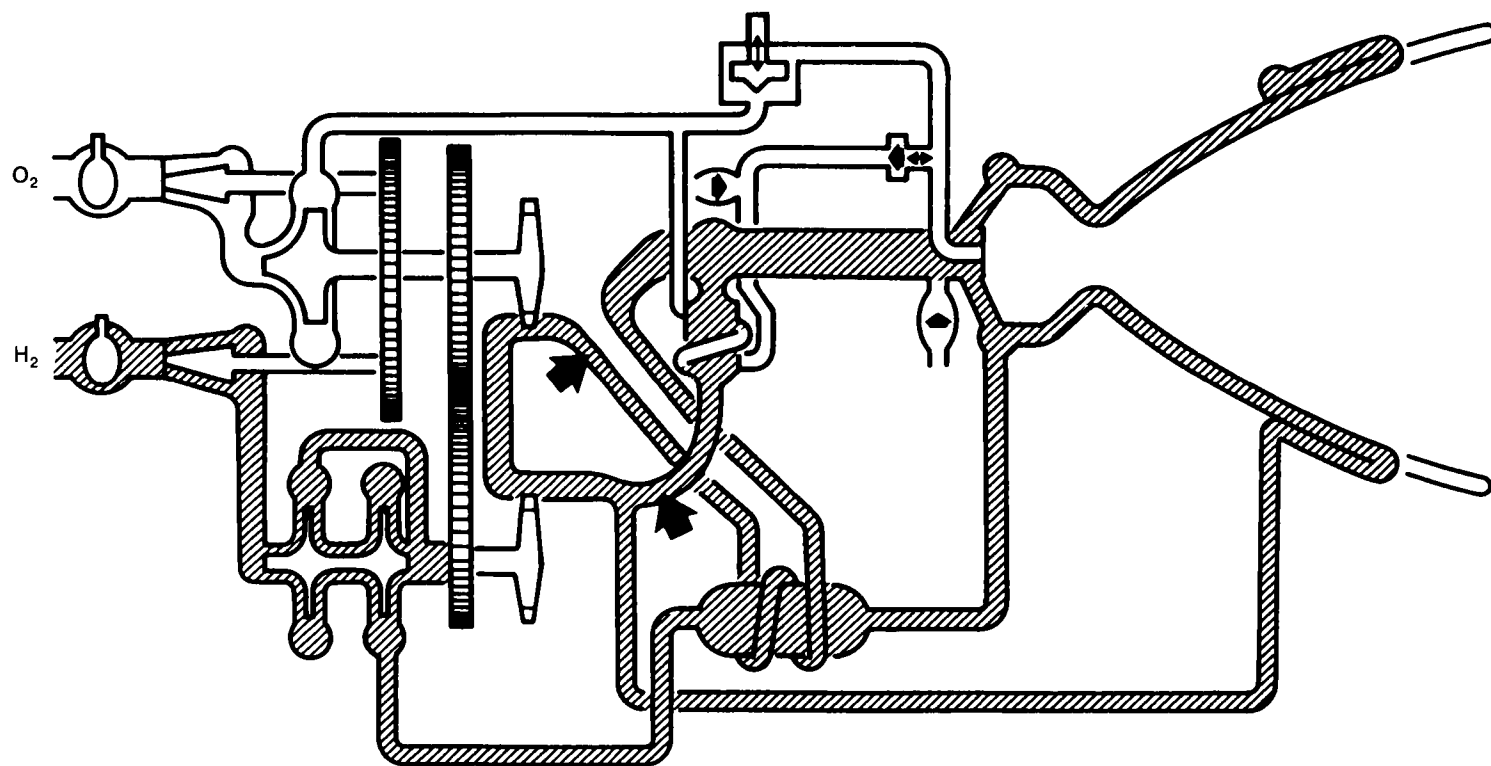
The advanced expander cycle OTV engine (as defined under Contract NAS8-33567) was the baseline engine used to incorporate full throttling capability. It was configured for operation at two low thrust levels in addition to its 15K full thrust level. The low levels (pumped idle at 1500 lb thrust and tank-head idle at approximately 100 lb thrust) were set based on the point design requirements and were not intended to be variable. If a high, continuous throttle range of 30:1 is required, revisions to the cycle flow paths must be made. The baseline cycle schematic is shown in Figure 6-1. The principal problem to be solved in making the engine fully throttleable is that of ensuring stability and efficiency (performance) over the full range of interest. (Based on a turbomachinery analysis, pump stability does not appear to be a significant concern — Figure 6-2)

Two concepts have been identified which appear to provide stable combustion over large throttle ranges. These concepts are the dual-area orifice injector and the  $\text{GO}_2$  injector. (Hydrogen injector throttling in an expander cycle engine is not normally a concern. Since the hydrogen is always a very superheated gas whose density is decreased with reduced throttle settings, only if the injected  $\text{H}_2$  temperature fell below approximately 200°R would combustion instability be a concern.)

The dual area orifice concept involves the addition of a second oxidizer injector plate and manifold and an appropriate shutoff valve for the secondary flow at the lowest thrust levels. This concept, as shown in Figure 6-3, has the advantage of being tested with  $\text{LF}_2/\text{GH}_2$ . (Final Report of High Energy Advanced Throttling Concept Study, AFRPL-TR-67-140) A throttle range of 170:1 was demonstrated, but there are weight and complexity penalties for an OTV engine application.

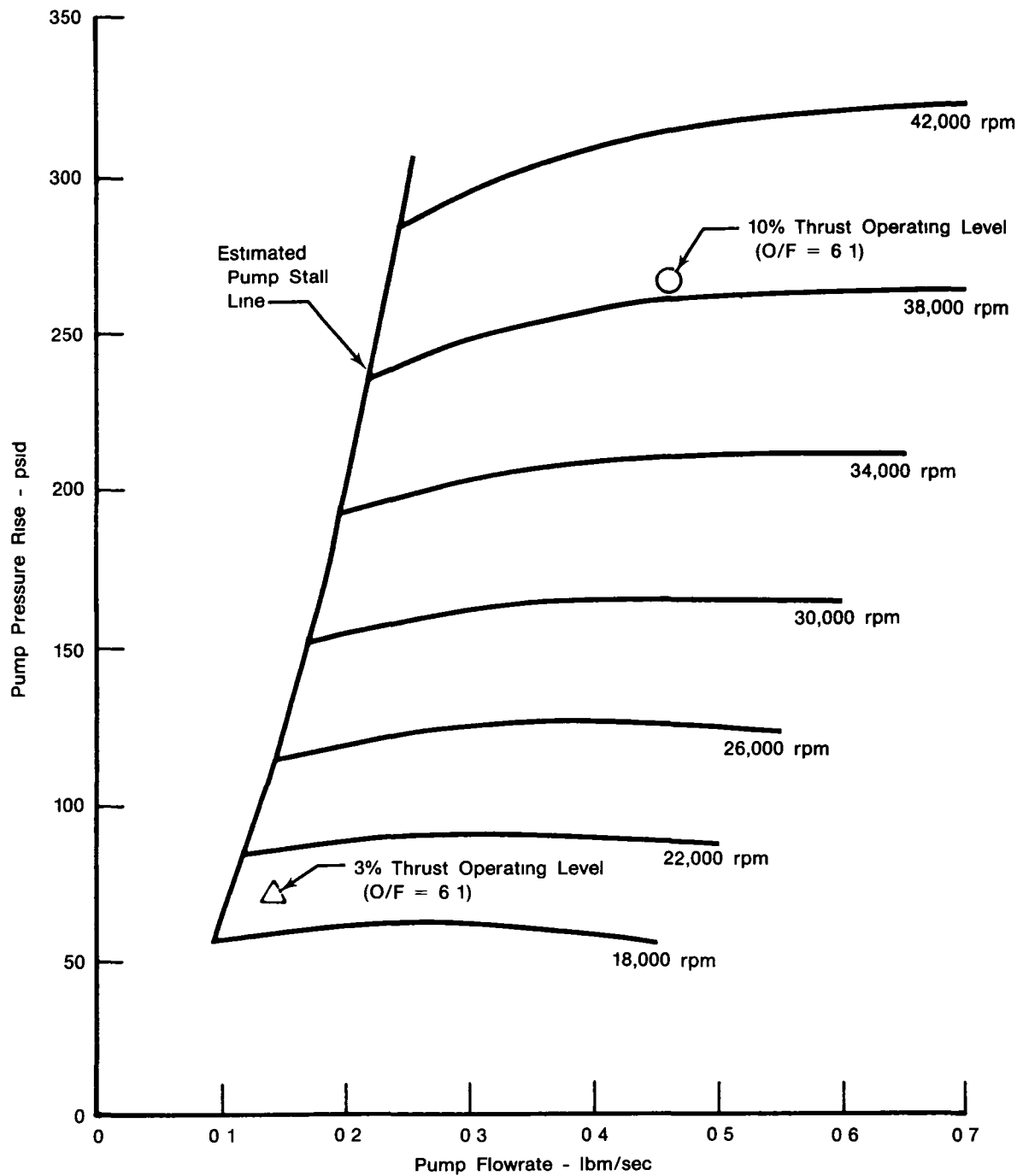
A similar concept using a dual-manifold, double-tangential entry slot oxidizer element was used in the XLR-129 engine preburner injector as illustrated in Figure 6-4.

The  $\text{GO}_2$  injection concept changes the  $\text{GO}_2/\text{GH}_2$  heat exchanger from the turbine bypass  $\text{H}_2$  line to the  $\text{H}_2$  regenerator discharge line, thereby utilizing all of the available hydrogen for oxygen vaporization. This concept, as shown in Figure 6-5, has the probable advantage of smoother throttling transients (no  $\text{GO}_2$  to  $\text{LO}_2$  phase changes) and good combustion performance, due to full  $\text{O}_2$  vaporization at all thrust levels, but has the disadvantage of not having been tested and it will be more difficult to design and fabricate the  $\text{GO}_2/\text{GH}_2$  heat exchanger than that of the baseline configuration.



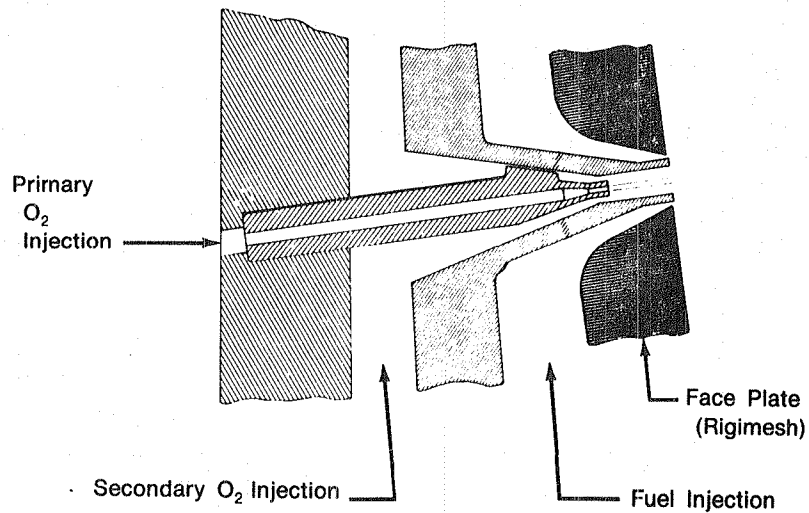
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Figure 6-1 — Advanced Expander Cycle Propellant Flow Schematic



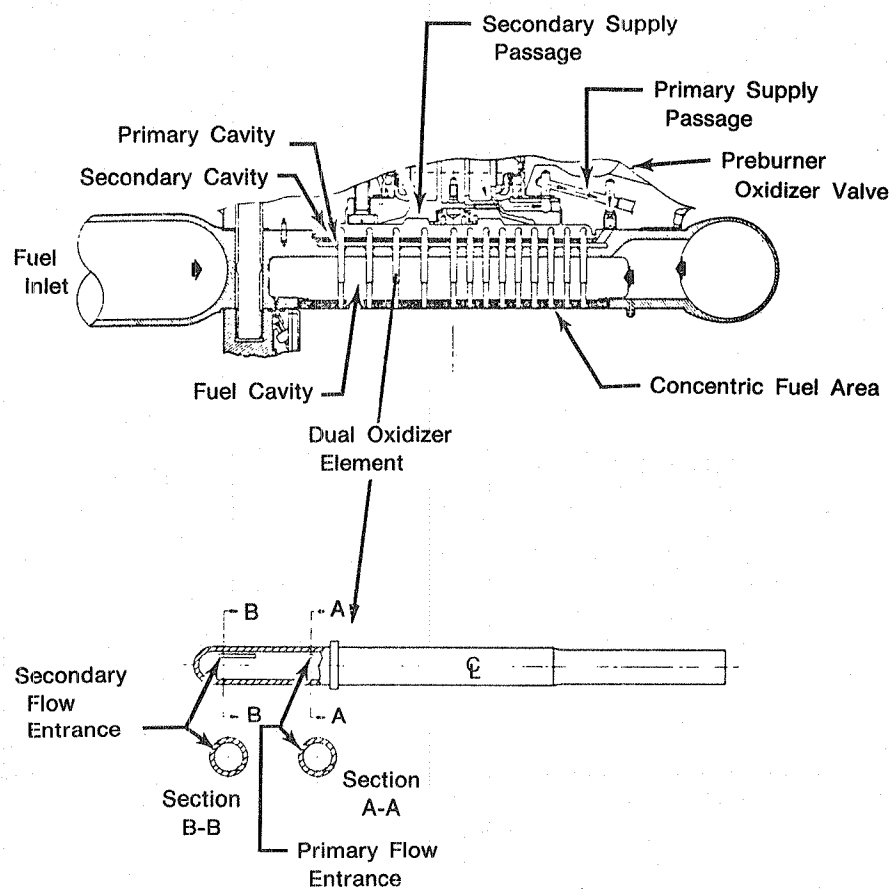
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Figure 6-2 — Advanced Expander Engine Fuel Pump Characteristics



FD 254300

Figure 6-3. — Dual Area Orifice Injector



FD 25970C

Figure 6-4. — XLR-129 Demonstrator Engine Preburner Injector

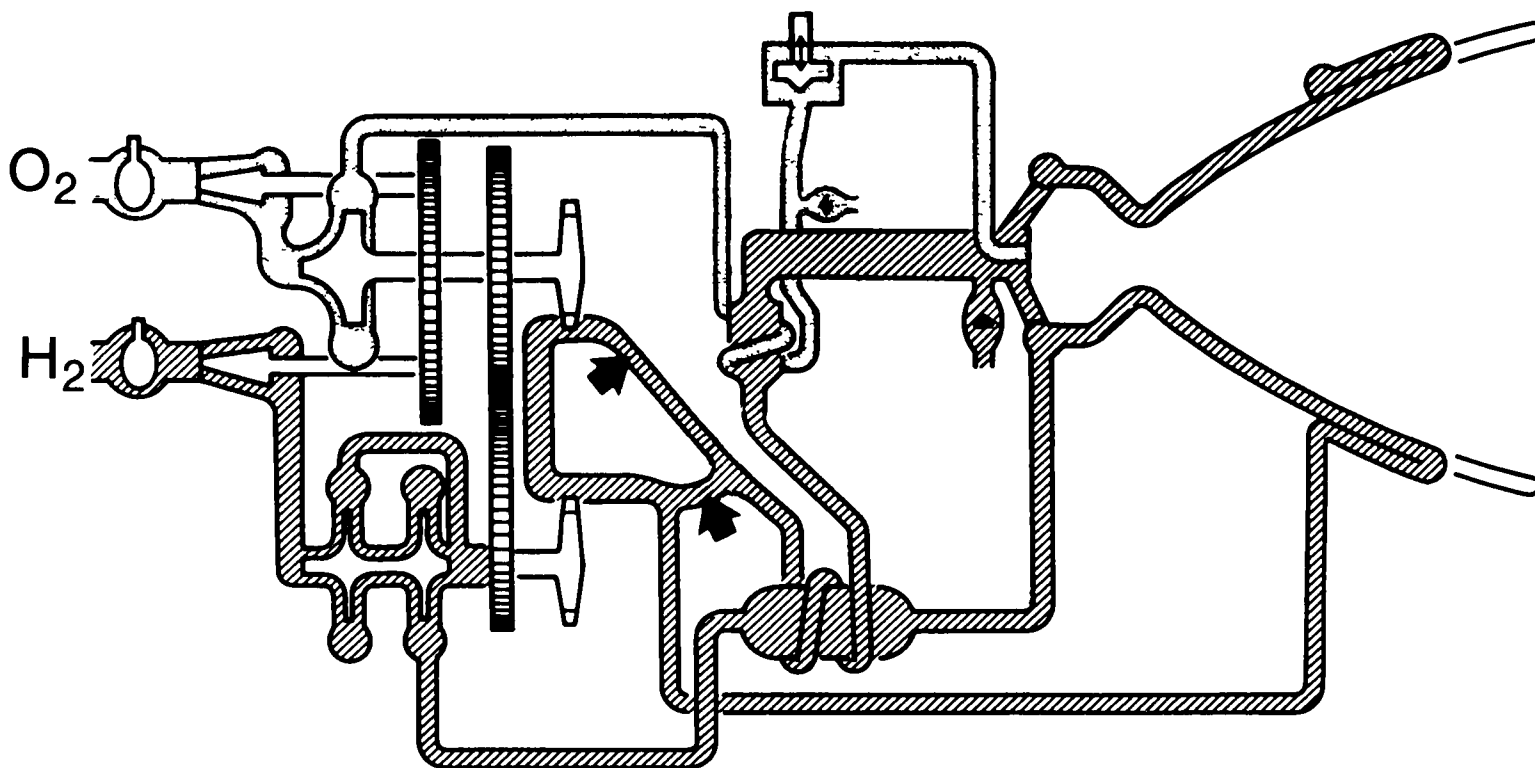


Figure 6-5 —  $GO_2$  Injection Concept

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Since combustion stability was not a problem in these designs and has been well-characterized, it appears that no problem exists in the development of such an injector, and further work is needed to determine the effect of these concepts on engine characteristics (e g , weight)

## SECTION 7.0 TECHNOLOGY EVALUATION AND RANKING

A methodology was established to allow an assessment of the critical technology items. The following discussion summarizes this methodology and presents the preferential ranking of the technology program elements.

### 7.1 METHODOLOGY DEVELOPMENT

Seven OTV engine characteristics are given in the SOW, as areas for technology improvement toward a specified goal. Improvement in each of these areas over the SOW baseline engine (the RL10A-3-3) will produce a benefit for a future Orbit Transfer System. However, these benefits will be in different areas. For example, improved specific impulse could provide an increased payload delivery weight capability or reduced operating cost due to reduced propellant requirements, whereas an increased engine life will require fewer engine overhauls and therefore a reduction in operational cost. In quantification of the benefit to the total program, several other factors become significant. Some of these factors are:

- Mission Type — High thrust versus low thrust delivery, manned versus unmanned, expendable versus reusable
- Mission Model — How many of each mission type over expected life of the OTV
- OTV Configuration — Vehicle length constraints, system NPSH capability, aerobrake versus all propulsive versus expendable vehicle
- Earth-to-orbit (ETO) Vehicle — Shuttle versus growth vehicle, cargo bay versus aft-cargo compartment, wet-launch versus dry launch

In addition to these factors, the OTV may be based at a Space Station/Space Operations Center. If this occurs, there will certainly be a significant impact on the OTV in areas such as maintainability, system condition monitoring, propellant requirements, and operational cost (e.g., \$/lb payload). The OTV engine characteristics and OTV benefits are summarized in Table 7-1.

TABLE 7-1 — ADVANCED OTV ENGINE CHARACTERISTIC BENEFIT INFLUENCE FACTORS

| <i>Engine Characteristic</i> | <i>OTV Benefit</i>               | <i>Also Affected By</i>                      |
|------------------------------|----------------------------------|--|
| 1 Specific Impulse           | Payload Weight, Operational Cost | Mission Type, ETO Vehicle, OTV Configuration |
| 2 Engine Weight              | Payload Weight                   | —  |
| 3 Life                       | Operational Cost                 | Mission Model, Mission Type                  |
| 4 Reliability                | Operational Cost                 | Mission Model, Mission Type                  |
| 5 Throttle Range             | System Flexibility               | Mission Model, Mission Type                  |
| 6 Zero NPSH                  | System Complexity                | OTV Configuration                            |
| 7 Engine Length              | Payload Geometry, Weight         | Mission Type, OTV Configuration, ETO Vehicle |

Note: Space station basing option also significantly affects benefits.

The latest available mission model generated by NASA-MSFC (Nominal High Energy Upper Stage Mission Model, Rev 6, dated October 1982) was used in the analysis and is summarized in Table 7-2

TABLE 7-2 — OTV MISSION MODEL  
SUMMARY

|  |      |
|--|------|
| Total program — 15 years (through year 2000)                 |      |
| Payload weight range — 3K GEO delivery to 13K GEO round trip |      |
| • Manned missions — 9  | ~ 5% |
| • Expendable (low thrust and planetary) — 29                 | 16%  |
| • Other missions (mostly GEO delivery/service) — 140         | 79%  |
| Total 178  | 100% |

Having established a data base for the OTV requirements, four separate rankings were generated. These were (1) payload delivery weight capability impact, (2) system life cycle cost (LCC) impact, (3) risk of achieving desired technology level within the necessary time frame (prior to 1990), and (4) criticality to the selected engine cycle design. In these rankings thrust chamber material, gears, bearings, seals, and turbopump shaft material technologies were used to increase the engine's life by increasing design margins rather than used to increase system pressure levels, rotor speeds, etc. This was done in keeping the Pratt & Whitney rocket design philosophy for high confidence in meeting operational reliability/life goals.

The rankings are discussed in the following paragraphs. (However, the engine capability to operate at zero NPSH during full-thrust burns was investigated for vehicle impact. This investigation did not reveal any positive benefits for an OTV vehicle system for operation at zero, rather than low NPSH. Therefore, the characteristic was dropped from further consideration.)

### 7.1.1 Payload Delivery Weight Capability

Of the items which affect payload delivery weight capability, engine specific impulse, and dry weight are the most obvious. Typical values for these are 85 lb payload/sec Isp and -2.6 lb payload/lb dry weight (on a typical LEO-GEO delivery, OTV round trip mission). Less obvious is engine length, however, this impact has been estimated for one specific application to be on the order of 200 lb of payload for each inch of length saved. (This factor is only useful for low density payloads which are not ETO vehicle payload mass capability limited.) Engine throttle capability is an enabling feature for low thrust transfer missions (i.e., the mission cannot be flown without an engine low thrust operating level available). In addition, recent studies have shown a low thrust mission benefit if the propulsion system can be throttled to provide a constant acceleration level to the payload rather than a fixed low thrust level (see for example, NASA CR-165277, "Primary Propulsion/Large Space Systems Interaction Study," Martin-Marietta Denver Aerospace, December 1981). These studies show that a gain of approximately 7 percent in payload can be attained (depending on acceleration level, number of perigee burns, etc.) if a continuous throttling capability is available. (There may also be a benefit to an aeroassisted vehicle if this capability is available, however this is very dependent on the aeroassist system design and could not be quantified for this study.)

These four effects were then numerically combined to provide a basis for ranking the technology concepts. The expression for this combination in terms of the impact of a given concept over the SOW baseline (the RL10A-3-3 engine) is as follows:

$$\text{Technology concept impact} = \text{Concept Values} = \partial \text{Isp} + \partial \text{Eng } W_t + \partial \text{Eng Length} + \partial \text{Throt Range}$$

where

$$\begin{aligned} \text{Isp} &= 85 (\text{Isp}' - 444) (0.8) \\ W_t &= 2.6 (290 - W_t') (0.8) \\ L_e &= 200 (70 - L_e) (0.1) \\ \text{Tr} &= 1050 (\text{yes/no}) (0.2) \\ (\text{Yes} &= \text{has full throttle capability} = 1.0 / \text{No} = \text{no throttle capability} = 0.0) \end{aligned}$$

In this expression, the last factor (i.e., 0.8 for Isp) is the estimate for a fraction of the total mission model affected by the particular characteristics. For example, 80 percent of the missions are estimated to benefit from higher Isp while only the 20 percent that are low thrust would realize some benefit from full throttling capability.

The results of the evaluations are shown in Table 7-3 and show a technology potential gain of up to 2816 points over the baseline engine for individual technology concepts. This of course does not imply that the RL10A-3-3 engine is of no value for an OTV application nor that an advanced technology engine can deliver up to 2816 lb of payload more than the baseline engine. It does indicate that, on a payload delivery capability basis, a nozzle extension system has a high payoff. Two other items are of interest in this data. The goal engine at 5832 points is considerably above the highest single technology concept and while still higher values could be reached by combination of various concepts, Pratt & Whitney believes goal engine's total characteristics are not attainable with the potential technology advances during this decade (prior to 1990).

TABLE 7-3 — IMPACT OF TECHNOLOGY ITEMS ON ENGINE PAYLOAD DELIVERY CAPABILITY

| Rank | Item                            | $I_{sp}$ (sec) | $W_t$ (lb) | $L_e$ (in) | $T_R$       | Technology Value |
|------|---------------------------------|----------------|------------|------------|-------------|------------------|
| —    | Goal Engine                     | 520            | 360        | 40         | 30.1 (Full) | 5,832            |
| 1    | Nozzle Extension System         | 482            | 467        | 40         | 10.1 (Step) | 2,816            |
| 2    | Advanced Material Turbopump     | 477            | 310        | 40         | 10.1 (Step) | 2,774            |
| 3    | Full Throttling Capability      | 474.7          | 356        | 40         | 30.1 (Full) | 2,761            |
| 4    | Improved Material Chamber       | 474.7          | 341        | 40         | 10.1 (Step) | 2,582            |
| 4    | Advanced Technology Gears       |                |            |            |             |                  |
| 4    | Advanced Technology Bearings    |                |            |            |             |                  |
| 4    | Advanced Technology Seals       |                |            |            |             |                  |
| 4    | Improved Material Pump Shaft    | 482            | 427        | 60         | 10.1 (Step) | 2,499            |
| 9    | Advanced Expander Engine (1980) |                |            |            |             |                  |
| 10   | RL10A-3-3 (SOW Baseline)        | 444            | 290        | 70         | (Fixed)     | 0                |

### 7.1.2 System Life-Cycle Cost

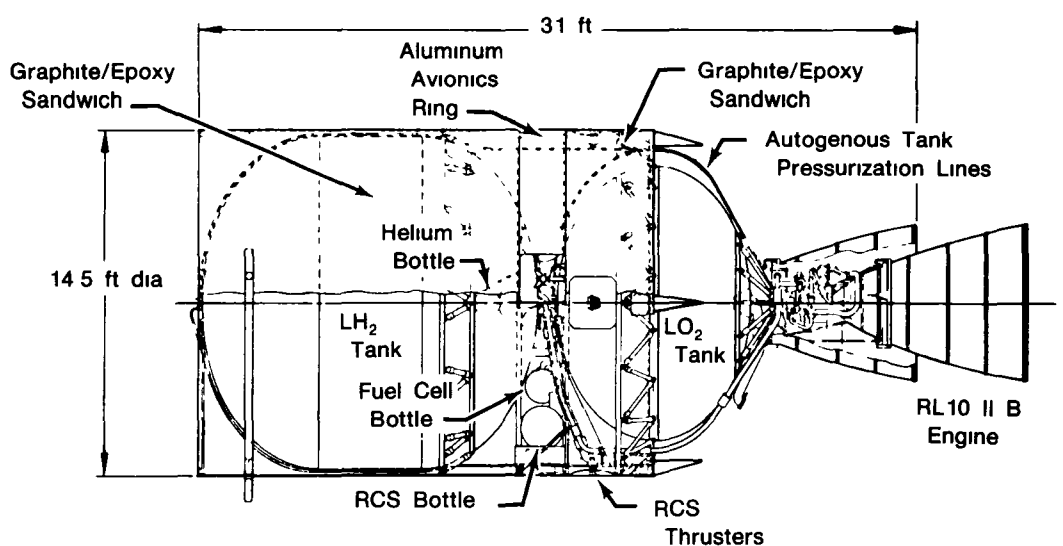
All changes in an engine's basic configuration might be expected to have some impact on the overall OTV program LCC. For example, a substantial improvement in payload delivery capability might well be expected to allow a reduction in total flights which should in turn reduce the total program costs (including the cost of technology development).

For cost spreading purposes, a baseline OTV program was selected. This baseline was a model generated by the Boeing Aerospace Company during their 1980 OTV Systems Study and covered a period of 18 years (Figure 7-1). The baseline program was then adjusted to include the AEE defined by P&WA in 1979 rather than the RL10 Derivative IIB engine utilized by Boeing.

It should be noted that the total program estimate of \$6 784B in fiscal year (FY) 1979 dollars becomes 16 85B in then year (TY) dollars using an 8 percent annual inflation rate

Baseline Cost - 1980 Boeing OTV Program Estimate

|            |  |
|------------|--|
| DDT&E      | — \$ 632 M (FY79\$)                            |
| Production | — 437 M  |
| Operations | — 5,715 M                                      |
|            | \$6,784 M (FY79\$) $\approx$ \$16,850 M (TY\$) |



FD 268852

Figure 7-1 — System Life Cycle Cost Baseline Program Model

The impact on the program cost of incorporating each of the eight technology items into the adjusted baseline was estimated. The single most significant cost factor in the operational part of the program was the cost of shuttle operations, and therefore a reduction in shuttle launches would be expected to result in a significant LCC benefit. The following assumptions were made in this regard:

- 1 Unlike missions would not be combined (manned, low thrust delivery, GEO delivery, and planetary missions cannot be mixed)
- 2 Military and civilian missions could not be combined
- 3 Commercial missions more than 4 months apart would not be combined due to loss of operating revenue

With these assumptions in mind, LCC estimates were made and the results are shown in Table 7-4. The highest benefit item (nozzle extension system) is principally due to a reduced number of shuttle flights. However, the total cost spread (reduced  $\sim 2\frac{1}{2}$  percent to increased  $\sim \frac{1}{2}$  percent) is so small that no clear case can be made for or against incorporation of any of the technology items.

TABLE 7-4 — LIFE CYCLE COST RANKING

| <i>Rank</i> | <i>Item</i>                      | <i>Program Cost<br/>(\$×10<sup>6</sup>)*</i> |
|-------------|----------------------------------|--|
| 1           | Nozzle extension system          | 16 44  |
| 2           | Improved gears                   | 16 83  |
| 3           | Improved bearings                | 16 84  |
| 4           | Improved seals                   | 16 84  |
| 5           | Baseline advanced expander       | 16 85  |
| 6           | Improved material pump shaft     | 16 85  |
| 7           | Improved material thrust chamber | 16 85  |
| 8           | Advanced material pump           | 16 91  |
| 9           | Full throttle capability         | 16 91  |

\* (FY79 \$ with 8% inflation)

### 7.1.3 Risk of Achieving Desired Technology Level

The objective of this study was to define technology concepts which would enhance the capabilities of an advanced OTV propulsion system. One of the principal assumptions of the study was that the advanced engine development program would begin in 1990. This assumption limits the period available to achieve the technology readiness of a concept for incorporation into an advanced engine design to approximately seven years.

Each of the technology concepts was evaluated considering the current state-of-the-art, the difficulty of the desired goal, the probable level of available resources and the available time for technology advancement. The results of this evaluation were combined as the risk of a particular concept reaching its desired goal. Since the advanced concepts are improvements over the baseline engine design (as described in Section 3.0), the risk identified is expressed as being greater than this baseline, which was defined as being compatible with 1980 state of the art. The results of this evaluation are summarized in Table 7-5 and are discussed in the following paragraphs.

TABLE 7-5 — ESTIMATED RELATIVE RISK OF TECHNOLOGY ITEM ACHIEVING ITS GOAL

| <i>Rank</i> | <i>Item</i>                      | <i>Risk</i>   |
|-------------|----------------------------------|---------------|
| 0           | Baseline advanced expander       | (Reference)   |
| 1           | Improved material pump shaft     | Very Low      |
| 2           | Improved gears                   | Low           |
| 3           | Improved material thrust chamber | Low-moderate  |
| 4           | Improved bearings                | Low-moderate  |
| 5           | Nozzle extension system          | Moderate      |
| 6           | Full throttling capability       | Moderate      |
| 7           | Improved seals                   | Moderate-high |
| 8           | Advanced material pump           | High          |

#### a Improved Material Pump Shaft

The improvement in the state of the art is significant relative to currently used materials for turbopump shafts. However, work in progress at P&WA on the development of materials for main shaft applications in advanced gas turbine engines promises to be of substantial benefit to the technology requirements of advanced rocket engines. This spin-off approach also implies

that the necessary resources are relatively low. On this basis, the concept was judged to be a very low risk.

*b Improved Gears*

This technology area encompasses two basic design needs. No design data is available for hydrogen-cooled gears that define gear wear/life in terms of load variations. The RL10 gear system is certainly a valid data point but the impact of varying load parameters (e.g., hertz stress, pitch line velocity) with this configuration is not available and therefore extrapolation to other loading conditions, while an undesirable practice, has been necessary. Therefore, the first design need is to establish such a load variation versus wear relationship for a fixed test specimen configuration. The second design need is to use this relationship to evaluate advanced base materials, coatings, etc. for application in an advanced OTV design.

Since the definition of the load/wear relationship should be relatively straightforward, the baseline could be more optimized to somewhat reduce gear loads and since there are several viable candidates to improve wear, this technology was judged to be a low risk achievement.

*c Improved Material Thrust Chamber*

This technology area has the objective of improving and optimizing the base material of an OTV type thrust chamber particularly in terms of increased cyclic life properties. The approach involves the use of the RSR process currently in use at P&WA which has shown excellent results in a number of metal composite systems. Several copper based systems should be evaluated including NARLOY-Z.

Since this technology effort involves the application of an established technique to a new material system, the risk of achievement was judged to be somewhat greater than the two preceding concepts.

*d Improved Bearings*

The technology improvements needed in bearing capability are in the area of long life at high turbopump speed levels. The approach involves an evaluation of the total (i.e., rolling element, cage, and race) system including the materials and configuration of each. The demonstrated state of the art of bearing DN in a similar application is  $2.77 \times 10^6$  (XLR-129). The time available for the technology program is adequate. Based on the above considerations, this technology effort was judged to be of low to moderate risk.

*e Nozzle Extension System*

This technology area has the objective of producing a very short nozzle extension system with lightweight nozzle extension segments, including the extension/retraction mechanism. The goals of the current study indicate that a short retracted engine length is desirable. However, since the expander power cycle depends on heat picked up by the hydrogen in cooling the thrust chamber, some of the extension segments must be regeneratively cooled.

The technical challenge of this technology is in the area of producing a nozzle extension system which is multiple sectioned (at least one regeneratively cooled), is failure tolerant (including coolant leakage and actuator linkage), and is lightweight. The risk of achievement of this technology was judged to be of moderate level.

#### *f Full Throttling Capability*

The OTV engine system needs that must be demonstrated to achieve this technology capability are the operation of turbomachinery that can operate stably over the entire throttle range and a combustion system (injector and thrust chamber) that can provide a stable, high performance over the full throttle range. The valve and control system design must also be considered in the design of a fully throttleable engine. The individual elements of this capability appear to be relatively low risk but because of the multiple component aspect, the overall risk of achievement was judged to be of a moderate level.

#### *g Improved Seals*

The improvements needed in seal technology are in the areas of long life and low leakage performance with the objective of demonstrating an improved capability over that of the baseline design. The controlled gap seal arrangement was selected as the baseline design as it was considered to have the lowest technical risk. However, this design has a potentially higher leakage flowrate than a rubbing type seal, on the other hand, the rubbing seal has a potentially limited life. Therefore, the technical challenge is to demonstrate the technology of a seal design that can operate satisfactorily at the required high rubbing velocities, have a long useful life, and only permit a minimal leakage. Based on these considerations, the technology risk was judged to be in the moderate-to-high range.

#### *h Advanced Material Pump*

This technology area has the objective of demonstrating the design, fabrication, and test performance of a turbopump consisting entirely of nonmetallic materials. Because this is a significant departure from current rocket turbomachinery designs, this technology was judged to be a high risk concept.

### **7.1.4 Criticality to Selected Design**

As discussed in Section 3.3, the expander cycle engine's most critical components are the thrust chamber and the turbomachinery (especially the main fuel turbopump). These components are the cyclic life (the thrust chamber) and the duration life (the turbopump gears, bearings, and seals) limiting parts of the engine as well as being principally responsible for the engine's cycle power. Therefore, the technology for these components must be assessed as being more critical to the engine's basic design than ones which enhance the engine's overall performance, weight, or flexibility.

The resulting criticality ranking is presented in Table 7-6 and described as follows:

- The thrust chamber material technology is rated first due to the life limitation and cycle power limitation aspects of this component.
- Improved gears are highly ranked due to the limited life nature of the component and because the use of the synchronizing gear enables the utilization of a simple, open-loop control system.
- Improved bearings and seals are ranked lower than the previous two because, even though they are life-limiting components, either the improvements in the state-of-the-art are not a major advance (bearings) or alternative designs exist (seals).

- The nozzle extension system was ranked fifth primarily because it is a performance enhancing concept (improved performance) rather than an enabling concept
- The improved material shaft was ranked sixth because its primary impact is in the area of turbopump durability, and this concept was judged to be less significant to the AEE than the preceding concepts
- The full throttling and advanced material turbopump concepts are either improved flexibility and/or enhanced performance concepts and are therefore not critical to the basic engine design

TABLE 7-6 — RELATIVE CRITICALITY OF TECHNOLOGY ITEMS TO SELECTED ENGINE DESIGN

| <i>Rank</i> | <i>Item</i>                      |
|-------------|----------------------------------|
| 1           | Improved Material Thrust Chamber |
| 2           | Improved Gears                   |
| 3           | Improved Bearings                |
| 4           | Improved Seals                   |
| 5           | Nozzle Extension Systems         |
| 6           | Improved Material Shaft          |
| 7           | Full Throttling Capability       |
| 8           | Advanced Material Pump           |

## 7.2 OVERALL TECHNOLOGY ITEM RANKING

The individual rankings of payload delivery capability, risk, and design criticality were combined to form a single prioritized list of the eight advanced technology items (The LCC ranking was not used because a clear ranking on that basis could not be determined ) In this combination, the ideal technology item would be one which was highly critical to the engine design, had a high payload delivery payoff, yet had a relatively low risk of goal achievement The resulting list is presented in Table 7-7 and discussed below

TABLE 7-7 — OVERALL PRIORITY RANKING OF TECHNOLOGY CONCEPTS

| <i>Priority</i> | <i>Item</i>                      |
|-----------------|----------------------------------|
| 1               | Improved material thrust chamber |
| 2               | Improved gears                   |
| 3               | Nozzle extension system          |
| 4               | Improved bearings                |
| 5               | Full throttling capability       |
| 6               | Improved seals                   |
| 7               | Improved material pump shaft     |
| 8               | Advanced material pump           |

Based on

- Payload delivery impact
- Risk
- Criticality to selected engine design

The first two items (improved material thrust chamber and improved gears) were selected based on their high criticality to the cycle and low risk even though no performance advantage was utilized The third selected item is the nozzle extension system based on the combination of

high performance advantage, moderate criticality, and moderate risk. Improved bearings based on their moderate criticality and low risk were selected as the fourth priority technology item. The full throttling capability ranked fifth overall based on its performance enhancing ability and moderate risk even though it is of a low criticality. Improved seals, based on their moderate criticality were ranked sixth in priority followed by the improved material shaft which was low in both criticality and risk. Rated last was the advanced material pump which was low in criticality and high in risk.

### **7.3 PROPULSION SYSTEM SYNTHESIS**

In addition to the eight technology areas which were ranked in the preceding sections, a propulsion synthesis activity is also required. This activity is necessary to provide the OTV program analysts with updated information on engine capabilities as well as to assess and modify, when necessary, the technology programs to reflect vehicle requirement decisions. The synthesis activity is summarized in the following paragraphs and should parallel any technology program activity.

#### **7.3.1 Incorporation of Technology Findings Into Engine Design**

The technology program elements are centered around some specific aspects of OTV engine technology. The individual elements must be brought together in an overall assessment in order to continue to have a viable engine design. For example, through the thrust chamber material program, if a significant increase in turbine inlet temperature becomes available, its impact on the engine cycle and the desirability of changes (e.g., increasing turbopump speed) must be evaluated in light of other technology program elements (e.g., gears, bearings).

#### **7.3.2 Updating of Engine Capability for OTV Program Analysis**

Whereas in the preceding section the overall engine analysis is necessary to coordinate the individual technology elements, it is also necessary to refine the engine capabilities for OTV program analysis. The impact of the technology program on the characteristics of the AEE must be presented to the OTV program analysts as relevant engine capabilities (e.g., specific impulse, weight, life, etc.) not as achievements in component technology (e.g., bearing DN, chamber hot wall temperature, nozzle weight, etc.).

#### **7.3.3 Evaluation of OTV Program Decisions**

At least three major OTV program decisions are likely to be made before the commitment to a full-scale OTV development. The decisions could have a major impact on the emphasis of engine technology.

- *All-propulsive or aeroassisted OTV* — In the current (and previous) study the OTV is assumed to be either all-propulsive or initially all-propulsion and evolve to aeroassisted with no engine change. If, however, the OTV were primarily aerobraked and were used only on certain missions (e.g., expendables) in an all-propulsion mode, the designs might be somewhat different in terms of thrust level, geometry, and specific impulse sensitivity.
- *Engine thrust levels and throttling requirements* — Previous studies have required the OTV engine to have a full thrust level in the 10-20K lb range, a pumped-idle level in the 1-2K lb range and a pressure-fed idle level. One of the objectives of this study was to add a 30:1 throttle range to the engine. If program decisions set a different full thrust level (e.g., 30-40K lb) or a different throttling requirement (e.g., fully throttleable between 500 to 3000

lb, step to full thrust), this impact on the engine cycle and on the technology program must be evaluated and changes made if necessary

- *OTV Basing* — OTV studies to date have generally assumed that the propulsion vehicle and its payload would be ground-based (launched by the shuttle for each mission) On some particularly demanding missions (large payload size and/or weight) this might be done on two launches with an orbital rendezvous and assembly performed in LEO The mission would then be carried out in a normal manner with the OTV returning to the shuttle for return to the ground With the detailed analysis now being conducted on the Space Operations Center/Space Station requirements, the impact on a space-based OTV needs to be addressed In the main propulsion area, if the system is to be space based and therefore returned to the ground as little as possible (ideally never), the engine must be initially designed for high reliability/reusability and for ease of turnaround maintenance, including component and/or complete engine change-out However, space-basing might well relax engine geometry constraints For instance, a long, fixed high-expansion-ratio nozzle might be used rather than a nested extension system

The typical current engine/propellant feed system interface has bolted flanges with ten or more bolts per line With two such lines, a four-bolt main gimbal attachment, two pitch/yaw actuator connections, a pneumatic line connection, four or more electrical power connections and instrumentation connections, change-out of such an engine in space would be very difficult particularly in an unpressurized shelter Therefore, an effort must be made in the initial design of an engine intended to be space-based, to minimize and greatly simplify the engine vehicle interface connections A similar problem is true for any engine component intended to be replaceable Because of the difficulty of space-based hardware change-out, engine reusability and inflight condition monitoring must be improved to minimize both the required maintenance and the precautionary change-out of questionable hardware

#### **7.3.4 Evaluation of Man-Rating Requirements**

It is probable that the single most significant consideration for the OTV propulsion system design is that of man-rating As of now, there are no hard ground rules for design criteria These ground rules must be developed and their impact understood prior to engine development For example, the standard fail-operational/fail-safe requirements may so complicate the engine system as to make mission reliability (as opposed to crew safety) unacceptably low It is even possible that a different engine system might be used on manned missions than that used on the more frequent unmanned missions Obviously, this topic needs much more analysis by both engine and vehicle system contractors working with OTV mission planners

## **SECTION 8.0**

### **ADVANCED TECHNOLOGY OTV ENGINE DESCRIPTION**

The results of trades involving performance, heat transfer, structural assessment, fabrication, component geometry, and weight requirements were considered in updating the design of the AEE. This activity was principally concerned with the following major engine components: thrust chamber/nozzle assembly, fuel/oxidizer turbopumps, and the radiation-cooled extendible nozzle. The updated engine design is discussed in the following paragraphs.

#### **8.1 THRUST CHAMBER/NOZZLE ASSEMBLY DESIGN**

The thrust chamber and nozzle size and contours were determined during a previous study (NAS-33657). A chamber length of 15 in. and a chamber contraction ratio of 4:1 were selected as optimum for the AEE considering heat transfer and cyclic life limitations. The analysis of the thrust chamber and nozzle was performed at the engine off-design mixture ratio of 7:0 operating point because it provided the severest thermal conditions in the operating envelope. A schematic of the thrust chamber/nozzle assembly is presented in Figure 8-1. A maximum performance nozzle contour with an area ratio of 640:1 was chosen for the engine based on the length limitations and design point chamber pressure and mixture ratio. A radiation-cooled carbon-carbon composite two-section extendable nozzle was selected over a conventional single-piece extendible nozzle because of the very short installed length required of this engine.

##### **8.1.1 Thrust Chamber**

The advanced expander cycle combustion chamber, throat, and primary nozzle are of one-piece, nontubular, regeneratively cooled construction, shown in Figure 8-2. The curved combustion hot wall design was chosen over a flat hot wall design since it provided greater cyclic life. The heat fluxes experienced by the chamber are high and require the use of a high thermal conductivity material such as copper. The thermal fatigue properties of pure copper can be improved with only a slight reduction in conductivity by alloying with small amounts of other metals.

Thrust chamber cyclic life is a major consideration in the selection of an engine operating point. The cyclic life of the regeneratively cooled thrust chamber results from the large thermal strains that are introduced between the heated inner wall of the chamber and the cooler outer structural wall. The problem of evaluating thrust chamber cyclic life capability has been approached by (1) identifying the critical locations in the thrust chamber for analysis, (2) determining the cyclic life capability at those locations, and (3) making modifications to the chamber geometry and/or engine operation to ensure that the life requirements have been met.

Axial cooling passages are milled in the thrust chamber liner OD and the passages are closed with electrodeposited copper. A shell of nickel is then electrodeposited over the copper to act as the strength-carrying member and outer wall. The nontubular construction begins at the injector face and terminates downstream of the throat at an area ratio of approximately 6:1, where the heat flux is low enough to allow the use of standard tubular construction. These slots vary in width and depth along their axial length to achieve the desired local coolant flow velocities. Constraints, used during the sizing of the coolant passage, are as follows: a maximum coolant Mach No. of 0.40 to limit the pressure loss through the passage and a maximum hot wall temperature of 1700°R to give 1200 cycles to failure. To increase this to the goal of 2000 cycles (500 × factor of 4) an improved material must be used.

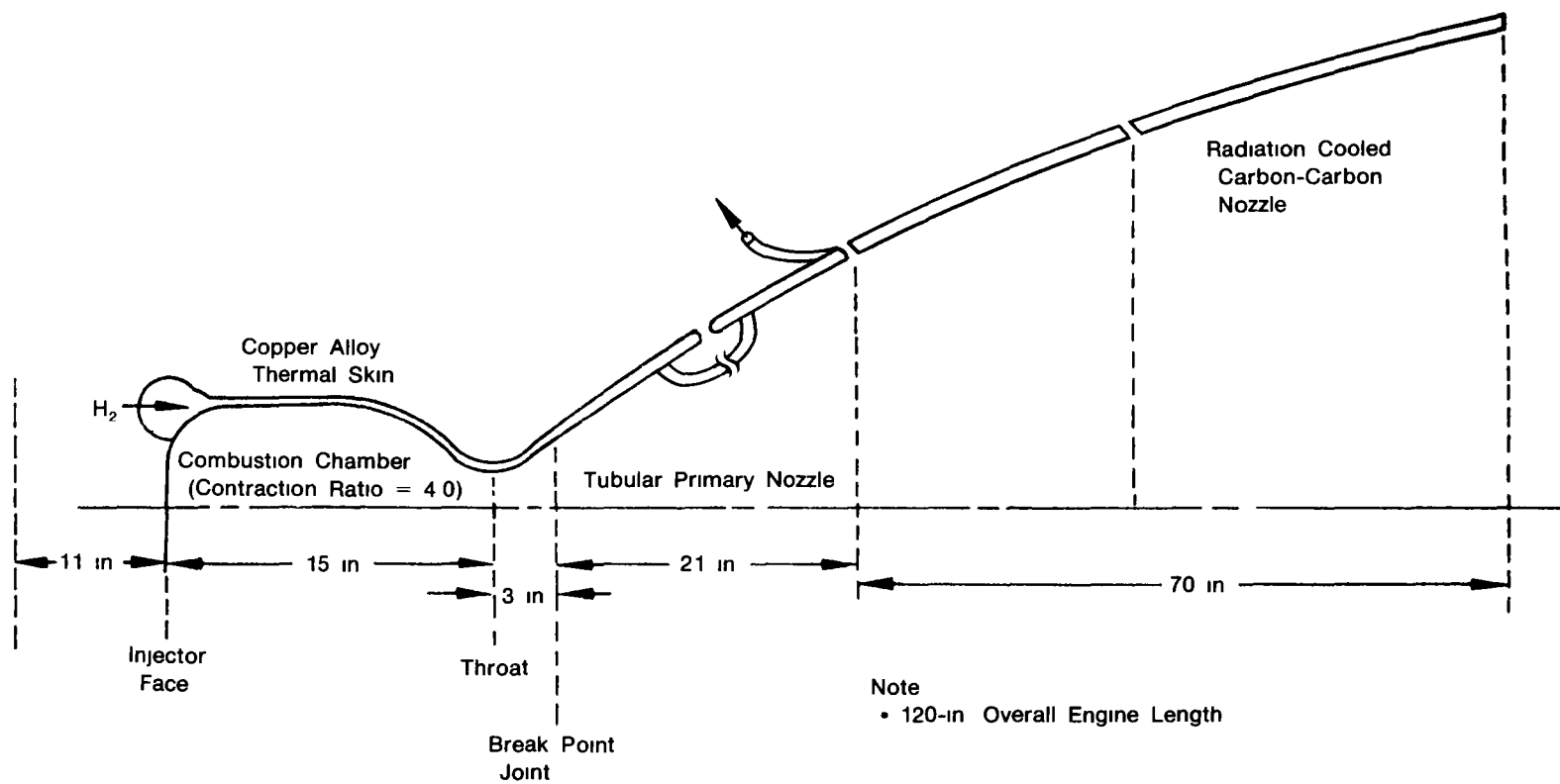
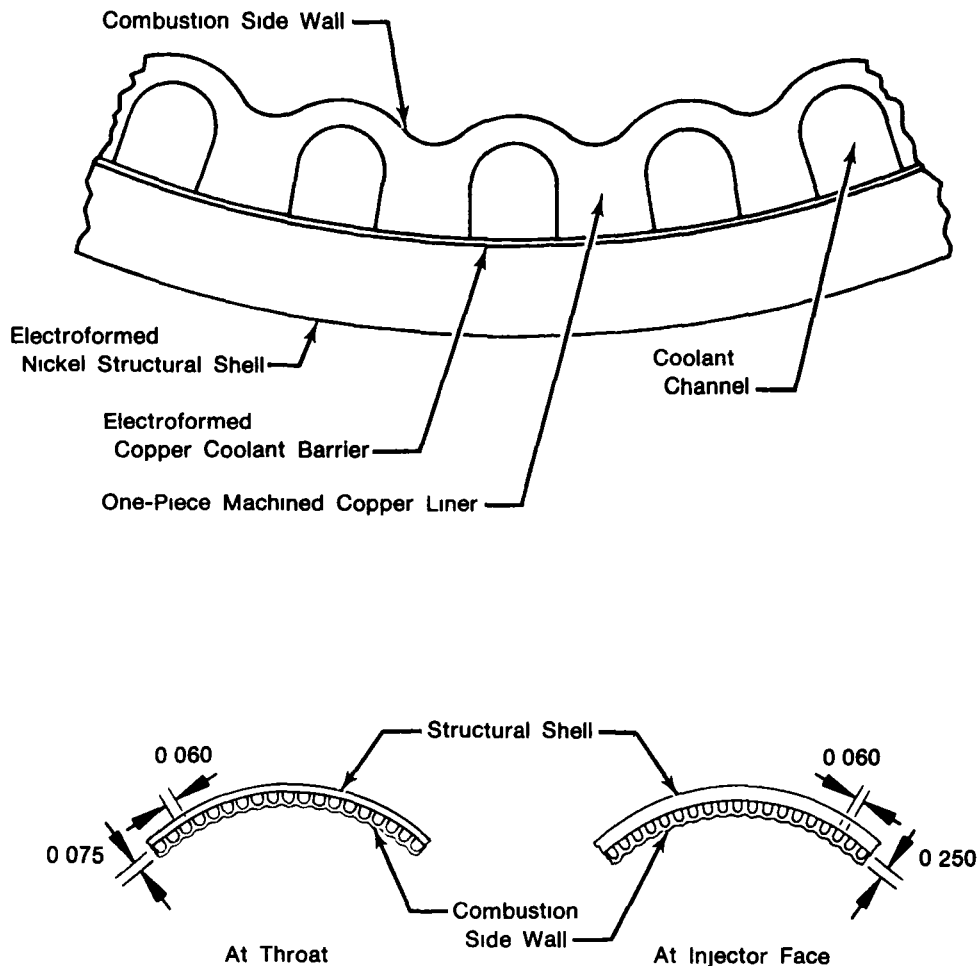


Figure 8-1 — OTV Engine Thrust Chamber and Nozzle Schematic



- Chamber Diameter 5.06 in  
 Throat Diameter 2.53 in  
 Axial Length 15.00 in
- 80 Milled Cooling Passage Slots Vary in Depth along Axial Length
  - Electro-Formed Outer Wall Provides Structural Shell

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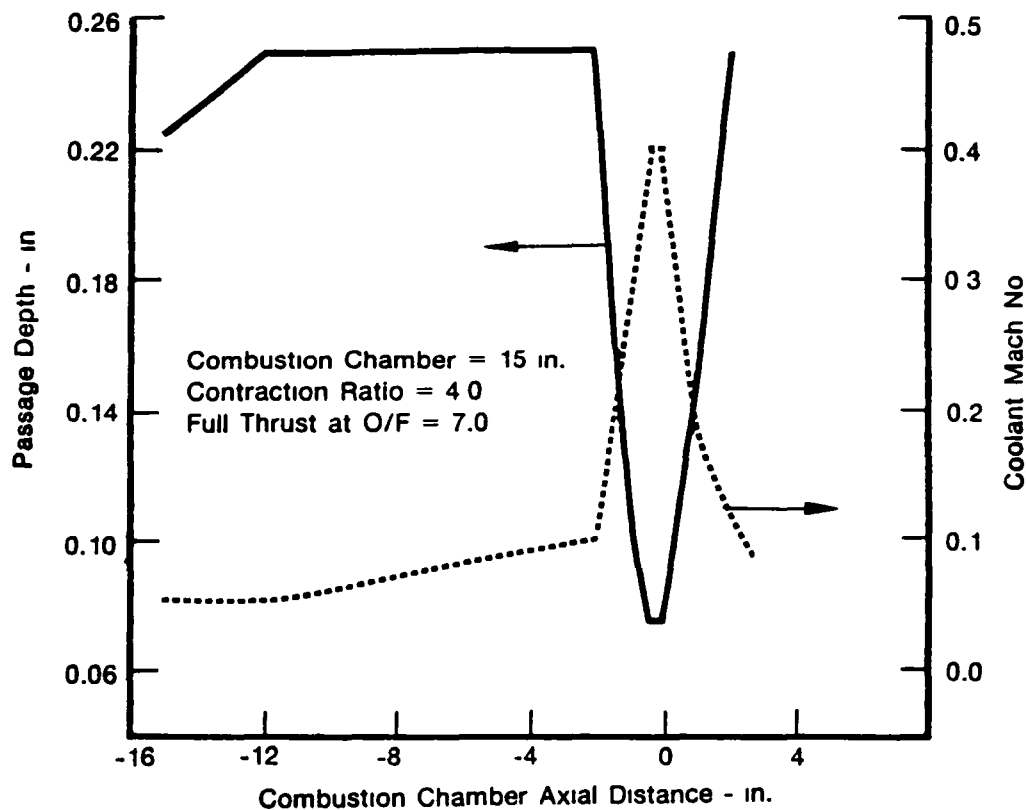
Figure 8-2 — Combustion Chamber Wall Detail

A contraction ratio of 4:1 was selected for the thrust chamber, based on previous parametric studies. The thermal analysis for the thrust chamber design was made at the off-design mixture ratio of 7:1 operating point, since it presented the severest conditions in the operating envelope. Parallel flow and counterflow cooling schemes were investigated for the chamber but the counterflow scheme was eliminated because of excessive pressure losses in the manifolding. Coolant passage geometry was defined and pressure and temperature characteristics were generated at mixture ratios of 6:1 and 7:1 for full thrust and at the pump idle and tank head idle design points. Table 8-1 presents these characteristics for the parallel flow

configuration with a full thrust counterflow point at an O/F of 7.1 included for comparison. The optimized passage geometry selected and all pertinent parameters for the mixture ratio of 7.1 design point are shown in Figures 8-3 and 8-4.

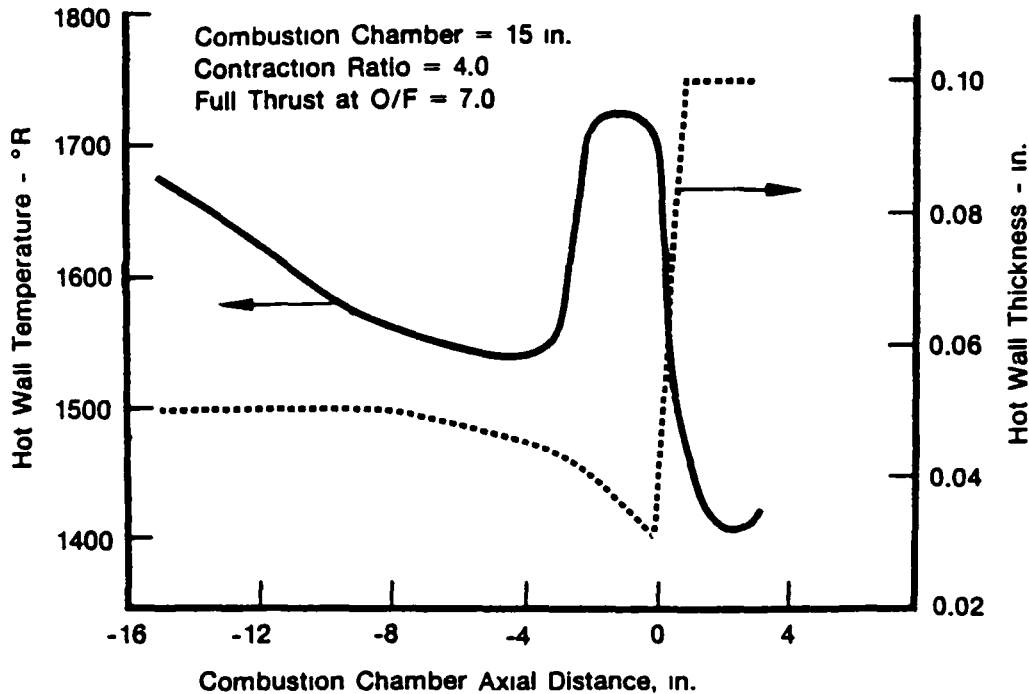
TABLE 8-1 — THRUST CHAMBER AND PRIMARY NOZZLE H<sub>2</sub> COOLANT CONDITIONS AT SELECTED DESIGN POINTS

|                             |                                | Parallel Flow |           |           |                |                       |
|-----------------------------|--------------------------------|---------------|-----------|-----------|----------------|-----------------------|
|                             |                                | O/F = 7.0     | O/F = 6.0 | Pump Idle | Tank Head Idle | Counterflow O/F = 7.0 |
| Thrust Chamber Thermal Skin | T <sub>in</sub> — °R           | 407           | 367       | 374       | 150            | 407                   |
|                             | P <sub>in</sub> — psia         | 4055          | 3859      | 304       | 16.2           | 4055                  |
|                             | T <sub>out</sub> — °R          | 845           | 738       | 876       | 543            | 844                   |
|                             | P <sub>out</sub> — psia        | 3673          | 3496      | 248       | 12.6           | 1561                  |
|                             | T <sub>hot wall max</sub> — °R | 1710          | 1514      | 1160      | 603            | 1460                  |
| Primary Nozzle              | T <sub>out</sub> — °R          | 990           | 874       | 1068      | 729            | —                     |
|                             | P <sub>out</sub> — psia        | 3597          | 3425      | 233       | 11.3           | —                     |



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Figure 8-3 — Combustion Chamber Coolant Passage Depth and Mach No



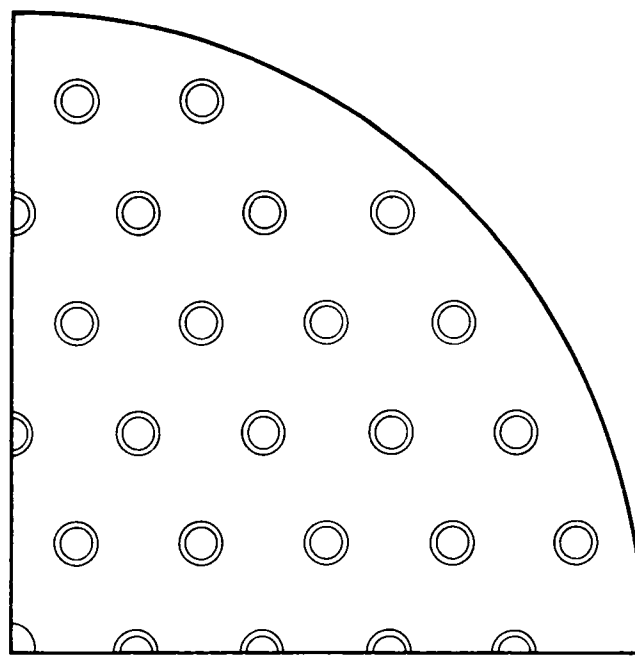
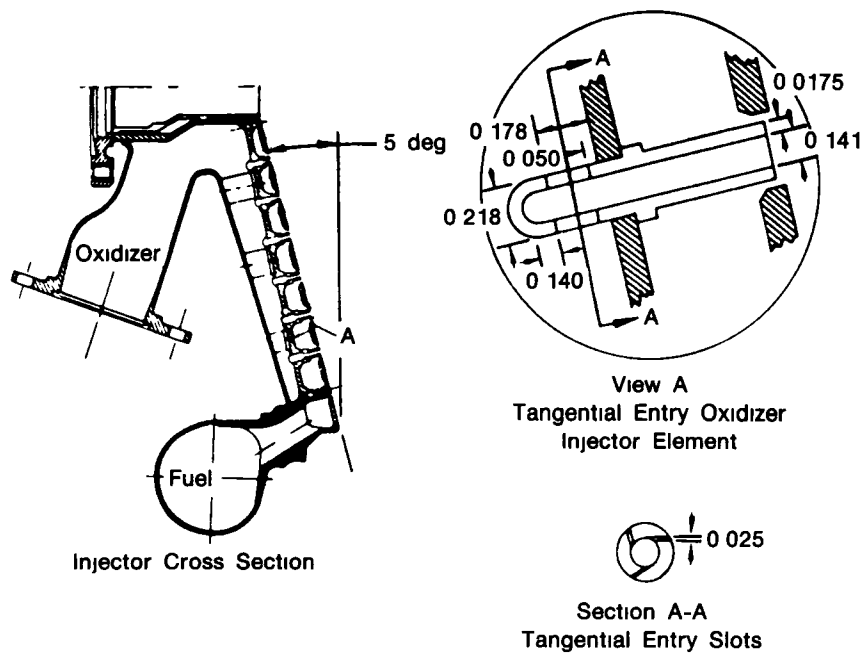
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Figure 8-4 — Combustion Chamber Hot Wall Temperature and Wall Thickness

### 8.1.2 Injector

The propellant injector is schematically depicted in Figure 8-5. The function of the propellant injector is to atomize the oxidizer and thoroughly mix fuel and oxidizer to provide the current conditions necessary for efficient combustion. The injector contains 84 tangential entry swirl injection elements arranged in a uniform hexagonal pattern around a central torch igniter. Liquid oxygen enters the injector through the oxidizer injector manifold, flows into the injector cavity and out oxidizer orifices into the combustion chamber. The oxidizer is admitted to the injector element through three tangential slots swirling the oxidizer flow and promoting mixing with hydrogen flow at the end of the element. The outer oxidizer elements of the injector are scarfed at a 45 deg angle to prevent oxidizer impingement on the wall. The injector face is coned at a 5 deg angle to prevent an oxidizer spud failure from producing an oxidizer impingement on the wall.

Gaseous hydrogen enters the peripheral fuel injector manifold and flows into the injector cavity. The fuel cavity has a 0.5 in height between the back of the injector faceplate and oxidizer cavity to minimize static pressure drop across the cavity, providing fuel flow uniformity. Most of the hydrogen flows out through the annular orifices around each oxidizer element. The full annular design of the fuel orifices is preferred for uniform distribution. It has extremely close tolerances and, since concentricity must be maintained, it may be necessary to insert three tangs into the annulus to preserve that concentricity.



- 84 Injection Elements
- Tangential Oxidizer Entry
- Transpiration Cooled Injector Face

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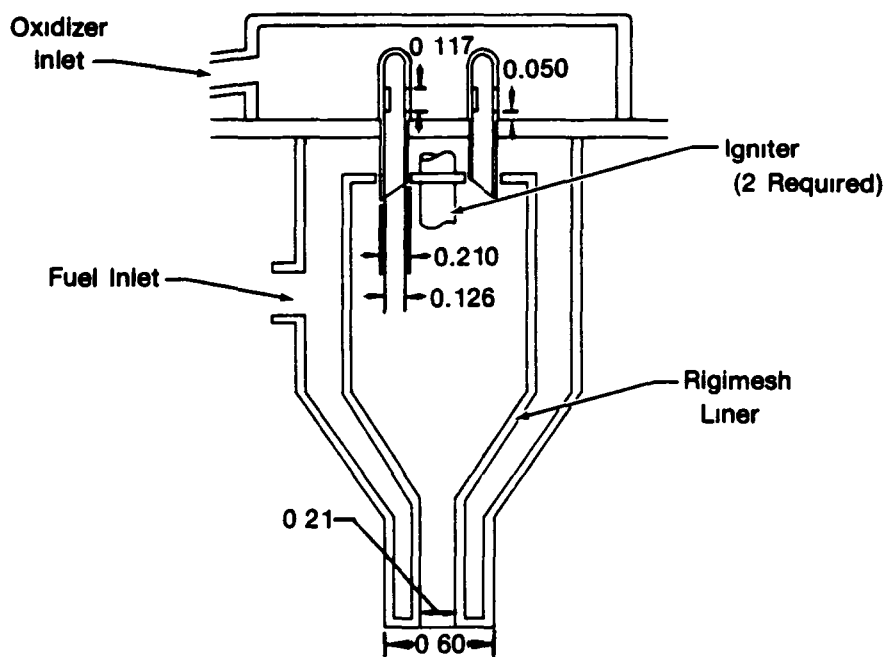
Figure 8-5 — Advanced Expander Cycle Engine Injector

The rigimesh injector faceplate uses a 400 standard cubic feet per minute (SCFM) rated rigimesh material to produce a cooling flow of 5 percent of the hydrogen flow at the design point. Standoffs required to attach the rigimesh are cylindrical, with as small a diameter as feasible. They are located equidistant from the three closest oxidizer spuds and are uniformly distributed at equal radii. The fuel manifold has a 0.50 in. height between the back of the rigimesh and the oxygen manifold. This separation is required to minimize static pressure drop between the outer and inner radii allowing optimum fuel flow uniformity.

Immediate contact between oxidizer and fuel is made at each element as the propellants leave the injector face and enter the combustion chamber. This configuration is designed to provide thorough combustion, high combustion efficiency, and high specific impulse.

### 8.1.3 Torch Igniter

Thrust chamber ignition is provided by a torch igniter system as shown in Figure 8-6. The igniter is centrally located in the injector face. A metered flow of hydrogen and oxygen is mixed in an igniter chamber, ignited by a spark, and passed into the combustion chamber to ignite the main propellants. Increased reliability is accomplished by providing dual exciters and spark igniters and, with continuous operation, by eliminating the need for igniter propellant shutoff valves. The dual spark and exciter configuration provides a fail-safe energy source and designing the igniter to operate at rated thrust with oxidizer and fuel igniter flows eliminates the possibility of igniter damage due to valve leakage.



FD 212880

Figure 8-6 — Advanced Expander Cycle Engine Igniter Assembly

The fuel and oxidizer is ignited by a spark exciter assembly which provides a minimum of 20 sparks/sec at an energy level of 0.1 joules. The total oxidizer flow is injected into the igniter through two tangential entry swirl elements located at the upper end of the igniter chamber. Fuel flow is split. Part of the flow is delivered to a concentric slot surrounding each oxidizer

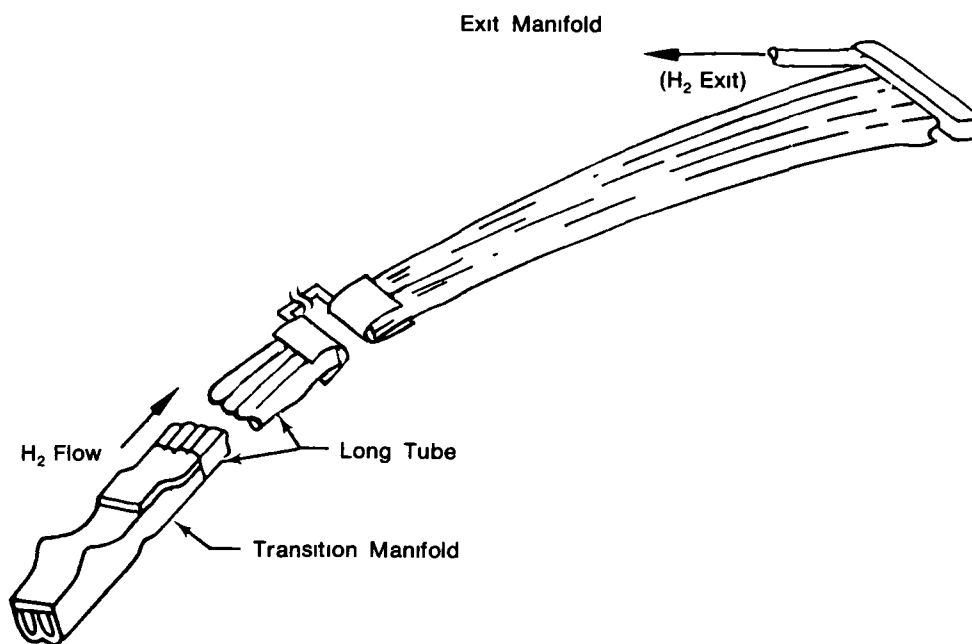
injector element and the remainder is used for igniter barrel cooling, flowing through a rigimesh liner

The rigimesh liner will allow continuous operation with high durability. The torch igniter is designed for continuous operation at an  $O/F = 4.1$ . This mixture ratio will burn cooler than the engine mixture of 6.0 and enhance the igniter life.

## 8.2 NOZZLES

### 8.2.1 Tubular Nozzle

In order to minimize the OTV engine length when stored in the space shuttle cargo bay, the engine length was shortened from 60 in to 40 in by the use of multiple extensions (see Section 5.0) to maintain the high area-ratio nozzle performance. The first section of the nozzle is tubular regeneratively cooled. It consists of a single-pass heat exchanger made up of two sections of 180 and 360 tubes, respectively, extending from the end of the nontubular section at  $x = 6$  to the start of the radiation-cooled nozzle. The two regeneratively-cooled sections are required to accommodate the short installed engine length. Flow is from the transition manifold at  $x = 6$ , parallel to the combustion gases through the first section tubes, through a connector manifold and then through the 360 tubes, to the exit manifold as shown in Figure 8-7.



FD 212857A

Figure 8-7 — Tubular Nozzle Schematic

Alloy PWA 770 (347SST) was selected as the tube material. The tubes are furnace brazed together to form a seal and are structurally supported by stiffener bands to carry the chamber hoop loads and minimize the effect of any flow-induced vibration. To establish band locations, tubes are treated analytically as beams subjected to thermal stress by the hot-cold wall temperature differential and from nozzle static wall temperature differential and bending stress from nozzle static wall pressure, longitudinal loads due to thrust, maneuver loads, and

gimballing acceleration are also considered. Bands are placed to establish beam lengths, which limit tube stresses to a level below the material yield strength at a factor of safety of 1.1.

### **8.2.2 Carbon-Carbon Extendible Nozzle**

A radiation-cooled carbon-carbon multiple extendible nozzle was selected because of its lightweight, favorable thermal characteristics, and short length. The thermal characteristics of the radiation-cooled secondary nozzle were defined at the severest thermal environment: full thrust and a mixture ratio of 7.1. A series of finger seals will be provided between the tubular primary nozzle and the nozzle extension sections. Maximum predicted nozzle temperatures were approximately 2400°R, which is well within the allowable temperatures for carbon-carbon material. Wall temperature profile characteristics are shown for the secondary nozzle in Figure 8-8.

### **8.2.3 Nozzle Extension System**

The nozzle extension actuation system, which was described in Section 5.0, was incorporated into the advanced engine design. The system consists of double-walled redundant actuators which allow the routing of the actuating gas to the actuators or retraction side of the piston.

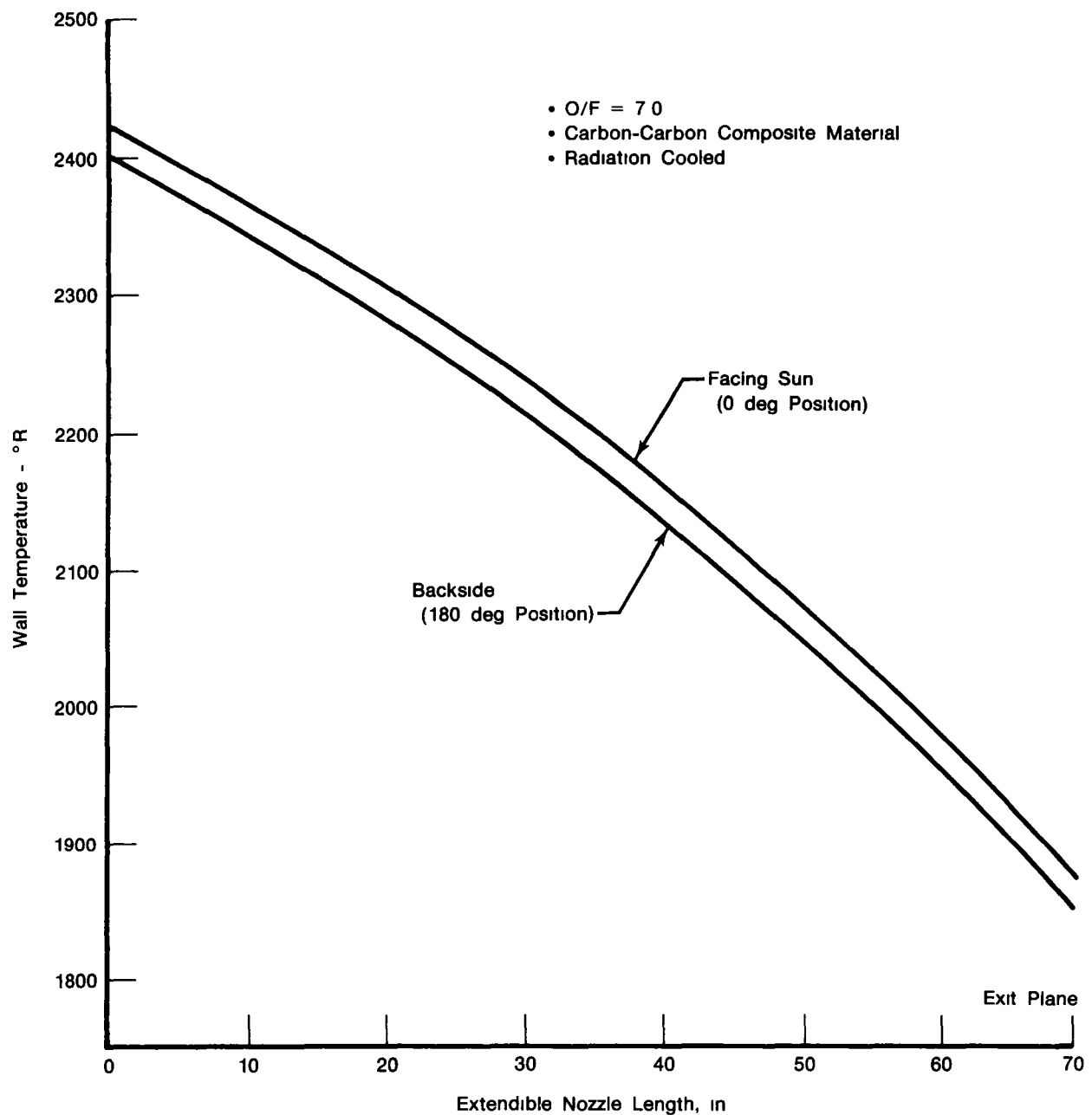
The nozzle is attached to the translating mechanism at three equally spaced points through a nozzle attachment bracket. The nozzle attachment bracket consists of a split circular ring and two-piece yoke. The gimbal attachment bracket provides 2 degrees of freedom to prevent transferring bending loads from the nozzle attachment.

## **8.3 HYDROGEN REGENERATOR**

The function of the hydrogen regenerator is to increase the turbine inlet temperature by recovering heat downstream of the turbines and by using it to preheat the fuel prior to cooling the thrust chamber and primary nozzle. This provides a higher fluid temperature at the turbine inlet, increasing the available turbopump power. Because of the relatively low thermal effectiveness requirements ( $\approx 40$  percent) of the regenerator, a cross-flow configuration was selected to provide ease of manifolding. The regenerator is a milled channel design consisting of a stack of 0.050 in. thick aluminum plates with small passages machined in each plate. Hot- and cold-side plates are alternated with the passages at right angles for a total of 61 hot and 60 cold plates. This design is lightweight, compact, easy to fabricate, and capable of withstanding the high hot- to cold-side differential pressure. Fluid and thermal analysis for the regenerator was carried out using a conventional effectiveness — number of transfer units (NTU) procedure. Figure 8-9 shows a sketch of the regenerator core arrangement and provides the design parameters and fluid condition at the design point.

## **8.4 GASEOUS OXYGEN HEAT EXCHANGER**

A GOX heat exchanger is required for the OTV engine to provide gaseous oxygen for propellant tank pressurization during full thrust and pumped idle operating modes and also to vaporize the engine oxidizer during tank head idle operation. However, studies indicate that a single compact heat exchanger could be subject to large, boiling-induced pressure, and flow oscillations on the  $O_2$  side at low mass qualities (less than 15 percent) which occur during the tank head idle mode.

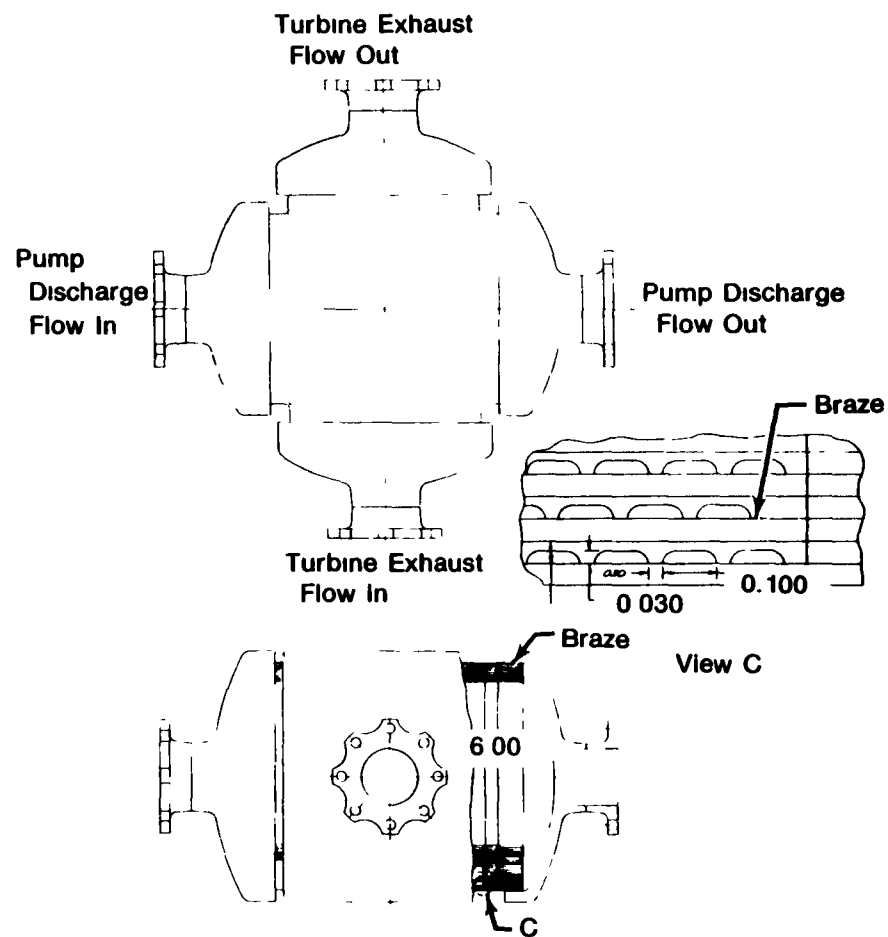


FD 212859A

Figure 8-8 — Extendible Nozzle Wall Temperature Prediction

This condition would result in unacceptable mixture ratio changes occurring in the main combustion chamber. The inclusion of a GOX vortex tube prevaporizer upstream of the GOX heat exchanger ( $O_2$  side) is recommended specifically to eliminate such a problem.

| Parameter  | Hot Side | Cold Side |
|--|----------|-----------|
| w (lb/sec)   | 4 22     | 4 35      |
| No. Plates   | 61.0     | 60 0      |
| No. Passages   | 2013     | 1980      |
| Ac (in. <sup>2</sup> )   | 7 32     | 7.20      |
| Dh (in.)   | 0.047    | 0 047     |
| q (psia)   | 1 66     | 0 34      |
| 4f (-)   | 0 025    | 0 025     |
| ΔP (psia)  | 6 71     | 1 47      |
| $h_c \left( \frac{\text{Btu}}{\text{hr ft}^2 \text{ } ^\circ\text{R}} \right)$ | 3783     | 3932      |
| $T_{in} (^\circ\text{R})$  | 797 0    | 97 0      |
| $T_{ex} (^\circ\text{R})$  | 503 0    | 367 0     |
| $P_{in} \text{ (psia)}$  | 1683 0   | 3895 0    |
| $P_{out} \text{ (psia)}$   | 1676 0   | 3893 0    |



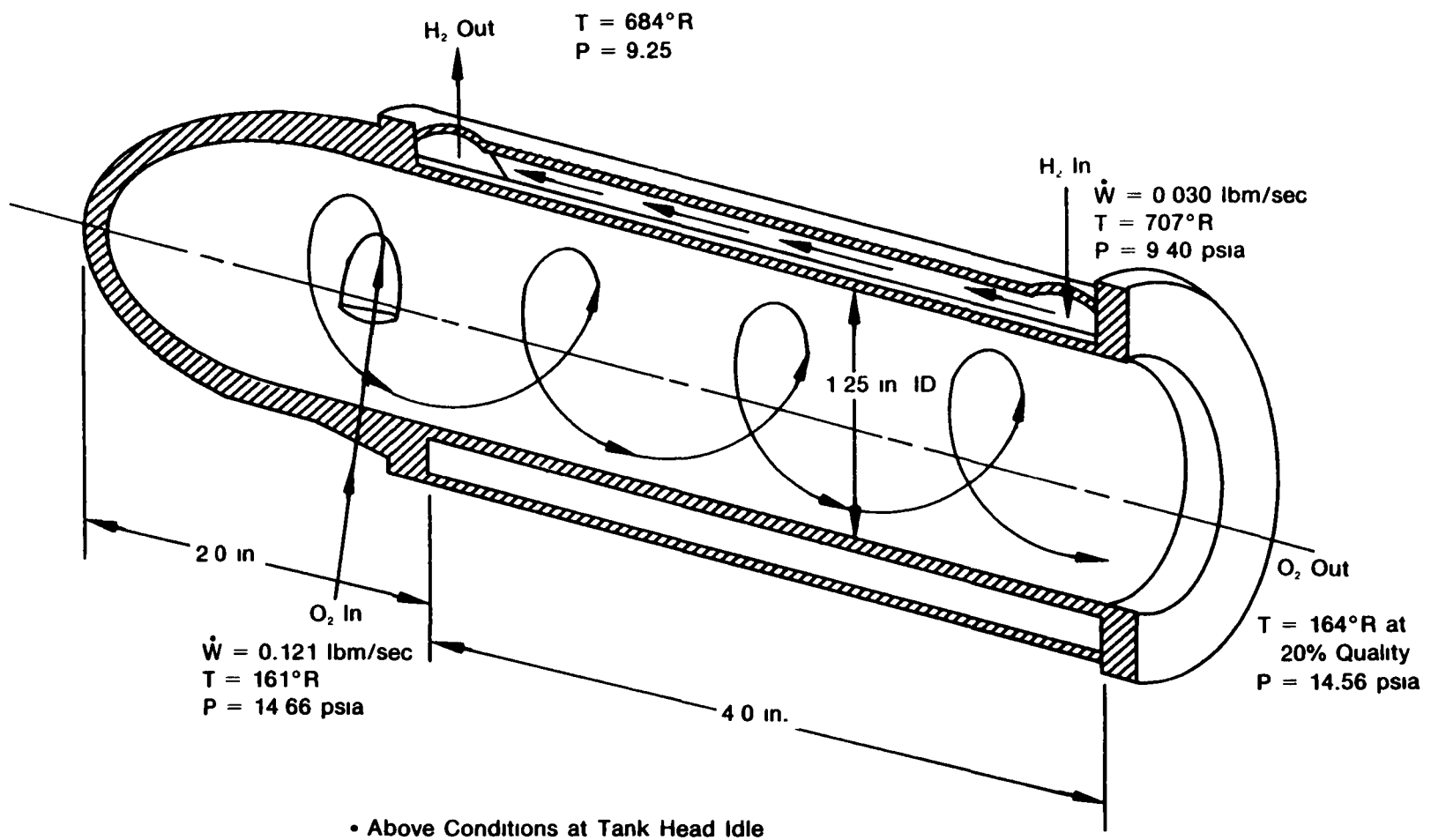
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Figure 8-9 — Single Pass Hydrogen Regenerator

The vortex tube prevaporizer concept is based on a unique application of state-of-the-art technology being studied for high-energy laser mirror and fusion target plate designs where high heat transfer rates and dynamically stable flow are critical requirements. The vaporization of liquid oxygen in zero g space environment is, furthermore, a logical application of the tangential entry, free vortex, swirl flow concept. The proposed design configuration of the vortex tube prevaporizer and tank head idle fluid parameters are shown in Figure 8-10 with the basic operation discussed in the following paragraphs.

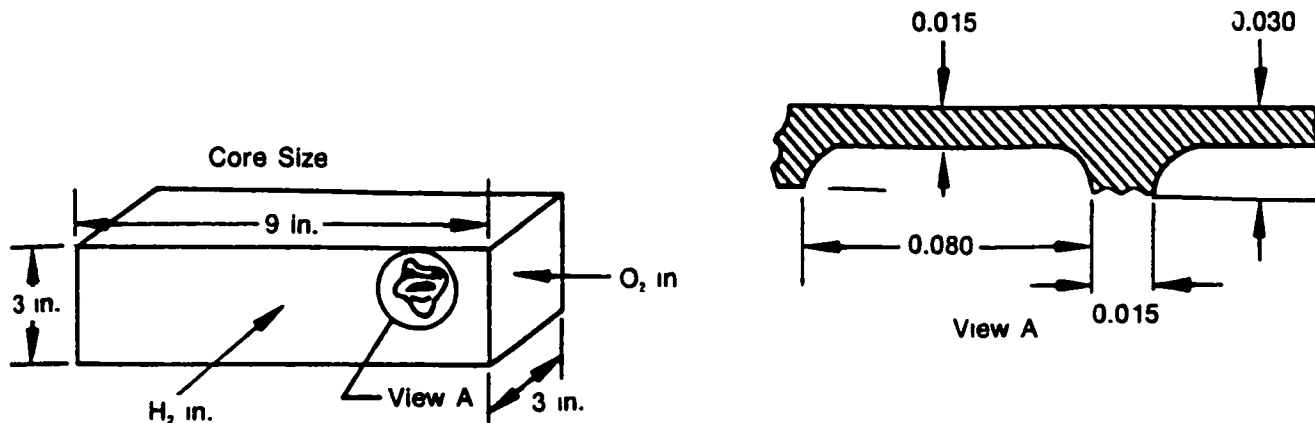
Saturated liquid oxygen (LOX) is injected tangentially near the closed end of a large diameter pipe and is allowed to spiral in a helical path toward the open end. The vortex pattern thus produced suppresses the transition from nucleate boiling to film boiling and allows extremely high heat transfer rates to be achieved. The centrifugal forces generated by the swirling flow force the liquid to the outer wall and allow the vapor to flow to the center of the tube. This action, in effect, separates the liquid and vapor phases so that boiling instabilities are not present. The liquid oxygen flowing along the wall is then allowed to stop its vortex flow pattern (by vanes or other antivortex devices) as it exits the prevaporizer, whereupon it flashes to tiny droplets and joins the vapor flow before entering the GOX heat exchanger. The heat source for LOX vaporization is the  $\text{GH}_2$  flowing in a jacket that surrounds the vortex tube.

The GOX heat exchanger was sized for single-phase gas conditions on both the hot and cold sides. The selected geometry was of a compact crossflow design utilizing a milled channeled construction. Figure 8-11 shows a sketch of the GOX heat exchanger core arrangement and provides important geometric and fluid flow design parameters at tank head idle and pumped idle.



FD 212862

Figure 8-10 — GO<sub>2</sub> Vortex Prevaporizer



Note: Sized for Gas Conditions Both Sides

| Parameter                             | Tank Head Idle | Pumped Idle |
|---------------------------------------|----------------|-------------|
| H <sub>2</sub> W (lbm/sec)            | 0.028          | 0.251       |
| O <sub>2</sub> W (lbm/sec)            | 0.128          | 2.783       |
| H <sub>2</sub> T <sub>in</sub> (°R)   | 882            | 940         |
| H <sub>2</sub> T <sub>ex</sub> (°R)   | 637            | 641         |
| O <sub>2</sub> T <sub>in</sub> (°R)   | 162            | 164         |
| O <sub>2</sub> T <sub>ex</sub> (°R)   | 612            | 218         |
| H <sub>2</sub> P <sub>in</sub> (psia) | 9.46           | 165.1       |
| H <sub>2</sub> P <sub>ex</sub> (psia) | 9.39           | 164.7       |
| O <sub>2</sub> P <sub>in</sub> (psia) | 15.60          | 205.4       |
| O <sub>2</sub> P <sub>ex</sub> (psia) | 15.53          | 193.6       |
| H <sub>2</sub> ΔP (psia)              | 0.07           | 0.4         |
| O <sub>2</sub> ΔP (psia)              | 0.07           | 11.8        |
| Q (Btu/sec)                           | 19.3           | 413         |

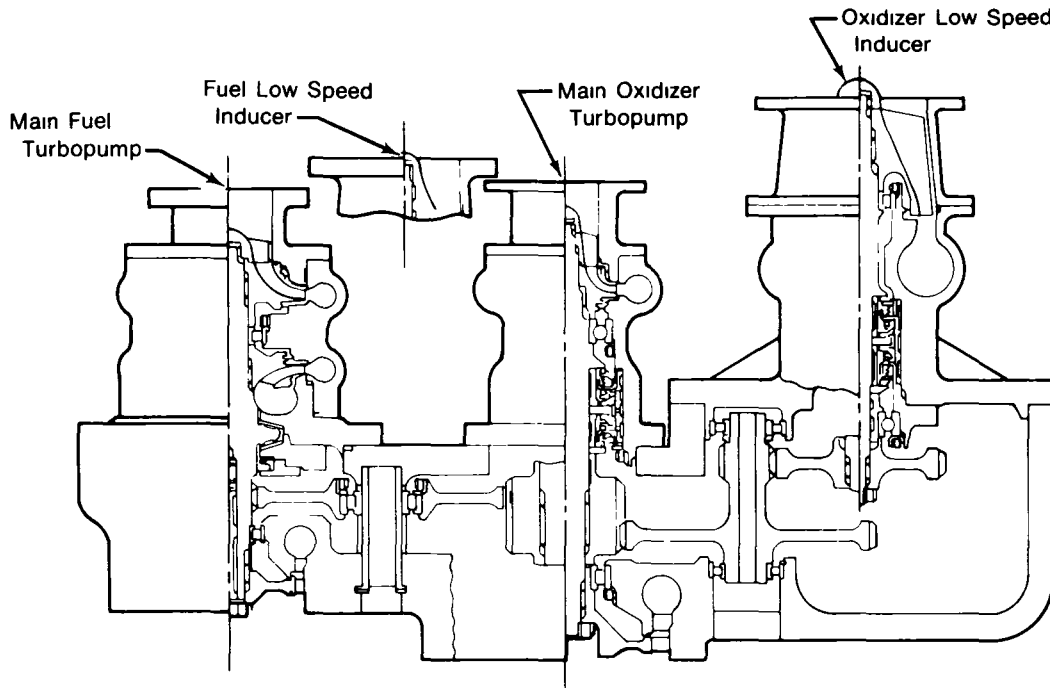
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Figure 8-11 — Gaseous Oxygen (GOX) Heat Exchanger

## 8.5 TURBOPUMPS

### 8.5.1 General Description

The fuel and oxidizer main pumps and low speed inducers comprise the turbopump assembly (Figure 8-12). The pumps are driven by the two single-stage turbines, a turbine on both main pump shafts. The low speed inducers are gear driven off the oxidizer pump shaft. A synchronizing gear has been included between the fuel and oxidizer main pump shafts to simplify the control system.



FD 212882

Figure 8-12 — Turbopump Assembly

The fuel pump is a two-stage centrifugal design driven by a single-stage turbine. The impellers, arranged back-to-back, are made of titanium and have diffusion bonded shrouds. The impellers and turbine disk are splined onto the shaft which transmits the torque. The pump housing is made of AMS 4215 aluminum (casing). The housing contains the two circumferential volute diffusers, each having a single conical discharge. The shaft is supported by two 20 mm roller bearings, one located between the two impellers and the other one immediately forward of the turbine disk. The bearings are hydrogen cooled and the aft bearing may be jetted with hydrogen if additional cooling is required. Tiebolts fore and aft on shaft maintain a preload on the impellers, bearings, and turbine. A double acting thrust piston has been incorporated onto the shaft to restrain the shaft thrust load. The piston is fed 2nd-stage impeller discharge pressure to each side. The resultant thrust is in the forward direction thereby allowing the aft piston pressure feed to be channeled back to the 2nd-stage inlet. The thrust piston lands rub against leaded bronze inserts in the pump housing. A controlled gap carbon circumferential seal is used to prevent the thrust piston, high-pressure hydrogen, from entering the gearbox cavity. The pump has the capability of being high-speed balanced as an assembly by the insertion of cylindrical weights into holes predrilled on the forward side of the impeller shroud and the aft side of the turbine disk.

The oxidizer pump incorporates several of the same features as the fuel pump (assembly balancing capability, single-stage turbine, volute diffuser). The single-stage shrouded centrifugal design is driven by a single-stage hydrogen turbine. The shaft axial thrust load ( $\sim 200$  lb) is restrained by a 25 mm ball bearing located just aft of the impeller. A 25 mm roller bearing is located on the aft portion of the shaft. The ball bearing is cooled by LOX while the aft roller bearing is cooled by hydrogen. A controlled gap, multiple vented cavity arrangement was used to prevent mixing of the hydrogen and oxygen. Two optional configurations for this seal are the rub face and hydrodynamic lift-pad types. All three types appear to provide adequate operational and life characteristics for the OTV engine application.

Both fuel and oxidizer low speed inducers are axial flow with three blades and three splitters. The inducers have an axial volute diffuser with a single discharge. The inducers are gear driven off the oxidizer pump shaft. A roller bearing has been placed under the inducer and a ball bearing just forward of the gear. The inducers are made of AMS 5362 stainless steel (SST).

The hydrogen flow to the two turbines is arranged in series, with the fuel pump turbine being upstream of the oxidizer turbine. The turbopump gears are encased in a one-piece aluminum casting gearbox. The one-piece gearbox minimizes gear misalignment by allowing the bearing races and mounting surfaces to be matched with a minimum of overall tolerances. The synchronizing gear and oxidizer-pump-to-fuel-low-speed-inducer idler gear have a single roller bearing to minimize misalignment. The gear teeth will be dry film lubricated. Spur gears, made of AMS 6265, are used exclusively with a diametral pitch of 18 and pitch line velocity of 39,300 feet per minute (ft/min). The gear train is lightly loaded and therefore can operate successfully at this high pitch line velocity.

The gears are cooled by the gaseous hydrogen that flows through the gearbox and is then used to cool the seal for the extendible nozzle. If it is determined that additional cooling is required, liquid hydrogen can be jetted onto the teeth. A hertz stress of 100,000 psi has been used to determine the gear teeth configurations. A reduction idler gear was needed between the oxidizer pump and oxidizer low speed inducer (LSI) to reduce the speed from 67,390 rpm to 9950 rpm.

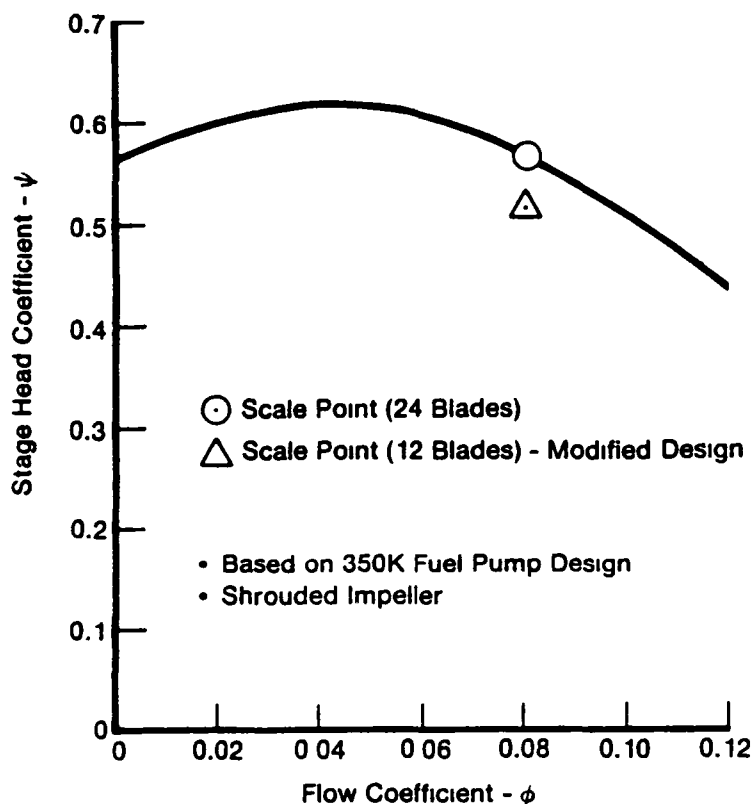
### **8.5.2 Main Fuel Pump**

The OTV main fuel pump is a two-stage centrifugal turbopump design. The design constraints which were established for this pump include (1) 2000 ft/sec tip speed limit for the bonded shrouded impellers with 25 deg back-swept blades, (2)  $3 \times 10^6$  DN bearing limit, and (3) 25 percent critical speed margin.

Sizing estimates showed that the pump would require a minimum bearing size of 20 mm. In the interest of attaining the highest possible specific speed and efficiency, the maximum allowable design rpm, based on the bearing DN limit ( $3 \times 10^6$ ), was established at 150,000 rpm. The shaft length was then sized to accommodate the two shrouded impellers plus inducer, two bearings, second stage volute inlet, thrust piston and the turbine rotor. This configuration just met the 25 percent critical speed margin requirement at the 150,000 design rpm. Both stages were designed for a resultant specific speed of 811.5. To obtain the necessary stage head-rise, a tip speed of 1960 ft/sec, slightly less than the 2000 ft/sec tip speed limit with a 3 in impeller diameter, was required.

The impellers for the two-stage fuel pump are scaled from previously proven P&WA turbopump designs with shrouds added to control leakages. The first-stage impeller excluding the inducer, is scaled from the first-stage of the 350K fuel pump, a design which demonstrated 95 percent hydraulic efficiency with use of shrouds. As a modification for the OTV design, 12

splitter blades are removed due to the small size of the impeller, leaving 6 full blades and 6 long splitter blades. Analysis indicates that this will result in an 8.7 percent decrease in head coefficient, which is accounted for in the design as shown in Figure 8-13. The first-stage is scaled at a flow and head coefficient compatible with the desired specific speed. This results in only a slight loss of stage efficiency as indicated in Figure 8-14.

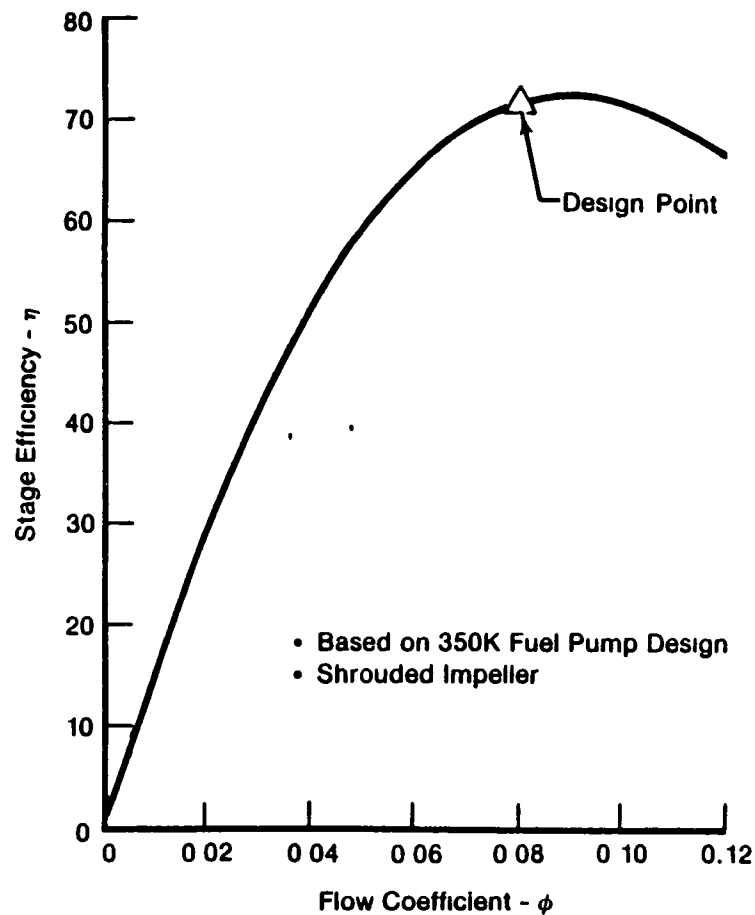


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Figure 8-13 — Fuel Pump First-Stage Impeller Head Coefficient

The second-stage impeller is scaled from the high efficiency second-stage of the XLR-129 fuel pump in the same manner as the first-stage. Again, the 12 short splitter blades are removed, resulting in a 10.3 percent loss in head coefficient as shown in Figure 8-15, leaving 6 full blades and 6 long splitter blades. The second-stage is also scaled at a flow and head coefficient necessary to obtain the desired specific speed, resulting in a minor efficiency loss as indicated in Figure 8-16.

Radial loads for each stage are found in Table 8-2 for the nominal design, off-design ( $O/F=7.1$ ), and pumped idle point. The impellers employ stepped labyrinth seals on the front and back shrouds at approximately 2 in. seal diameter to minimize leakage recirculations. Each impeller discharges into a constant velocity, single discharge volute collector followed by a conical diffuser.

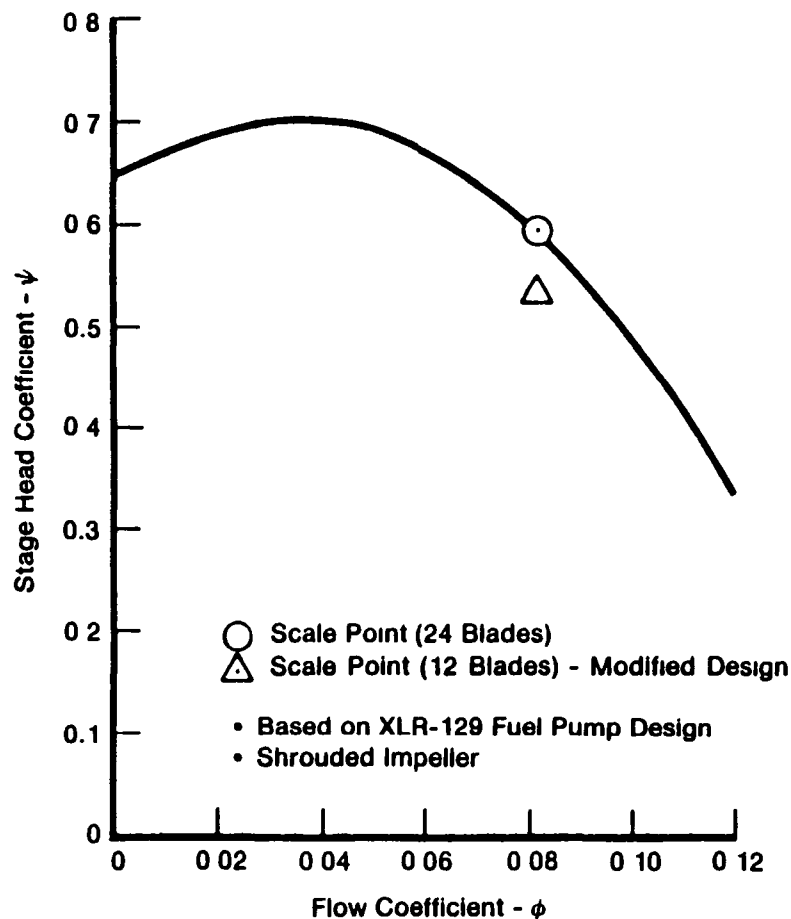


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Figure 8-14 — Fuel Pump First-Stage Impeller Efficiency

The pump configuration includes an inducer on the first-stage impeller to provide the required suction capability compatible with the fuel boost pump discharge, and ensure cavitation-free performance of the impeller. Three helical blades and a solidity of 1.5 were employed in the design providing a suction specific speed capability of 29,200 at an inlet tip flow coefficient of 0.013.

The overall pump efficiency is estimated to be 64 percent establishing a shaft horsepower requirement of 1571. To achieve this efficiency, the design will require tight seal clearances in order to minimize leakage recirculations. At least 80 percent volumetric efficiency is required with preliminary estimates indicating that this can be obtained by holding all diametral seal clearances on the impellers to 0.004 in. Mechanical and hydromechanical efficiencies were estimated at 94 percent and 85 percent respectively. Figure 8-17 shows a configuration drawing of the OTV main fuel pump.



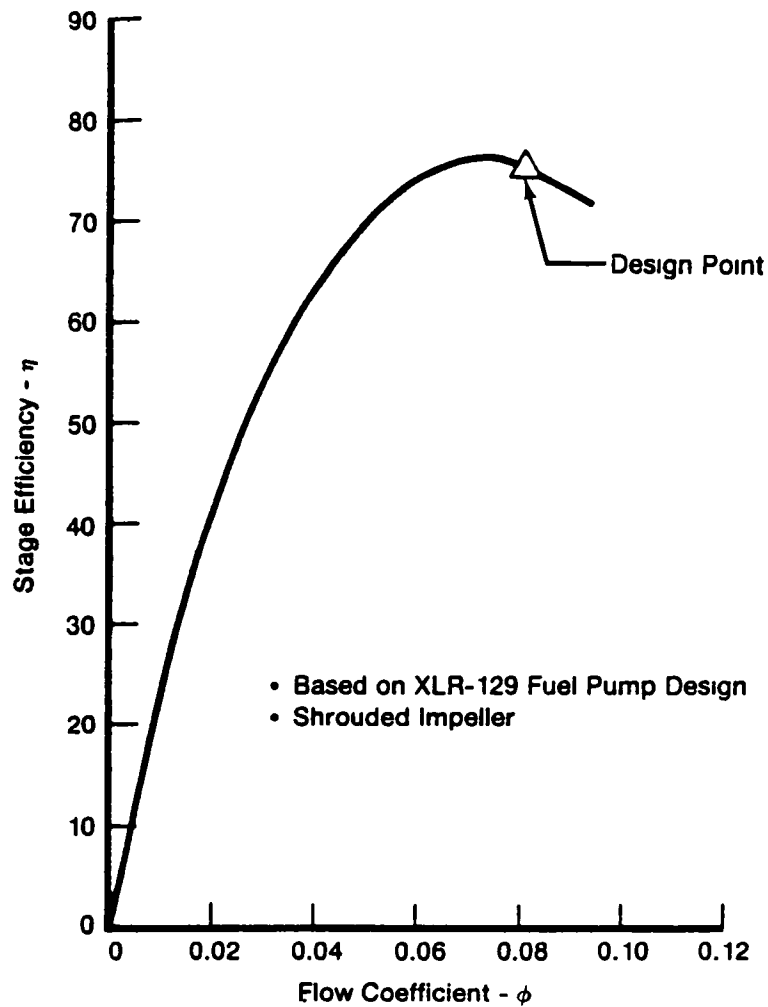
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Figure 8-15 — Fuel Pump Second-Stage Impeller Head Coefficient

### 8.5.3 Main Oxidizer Pump

The OTV main oxidizer pump is a single-stage, shrouded centrifugal turbopump design. The configuration consists of a three bladed inducer with solidity of 2.0, a shrouded impeller with 6 full length blades plus 6 long splitters, a constant velocity, single discharge volute collector, and conical diffuser. The pump impeller, a 25 deg backswept design, and inducer, are scaled from the P&WA space shuttle main engine (SSME) main LOX pump design modified by a slight extension in the impeller diameter to obtain the required head rise. The adjusted impeller diameter provides an optimized specific diameter for the design point specific speed of 1431.

The inducer is designed for a suction specific speed ( $N_{ss}$ ) capability of 23,000. The overall pump efficiency is estimated at 67.4 percent establishing a shaft-horsepower requirement of 375. As with the main fuel pump, the lab seal clearances will require close control to obtain the desired efficiency due to the small impeller size.

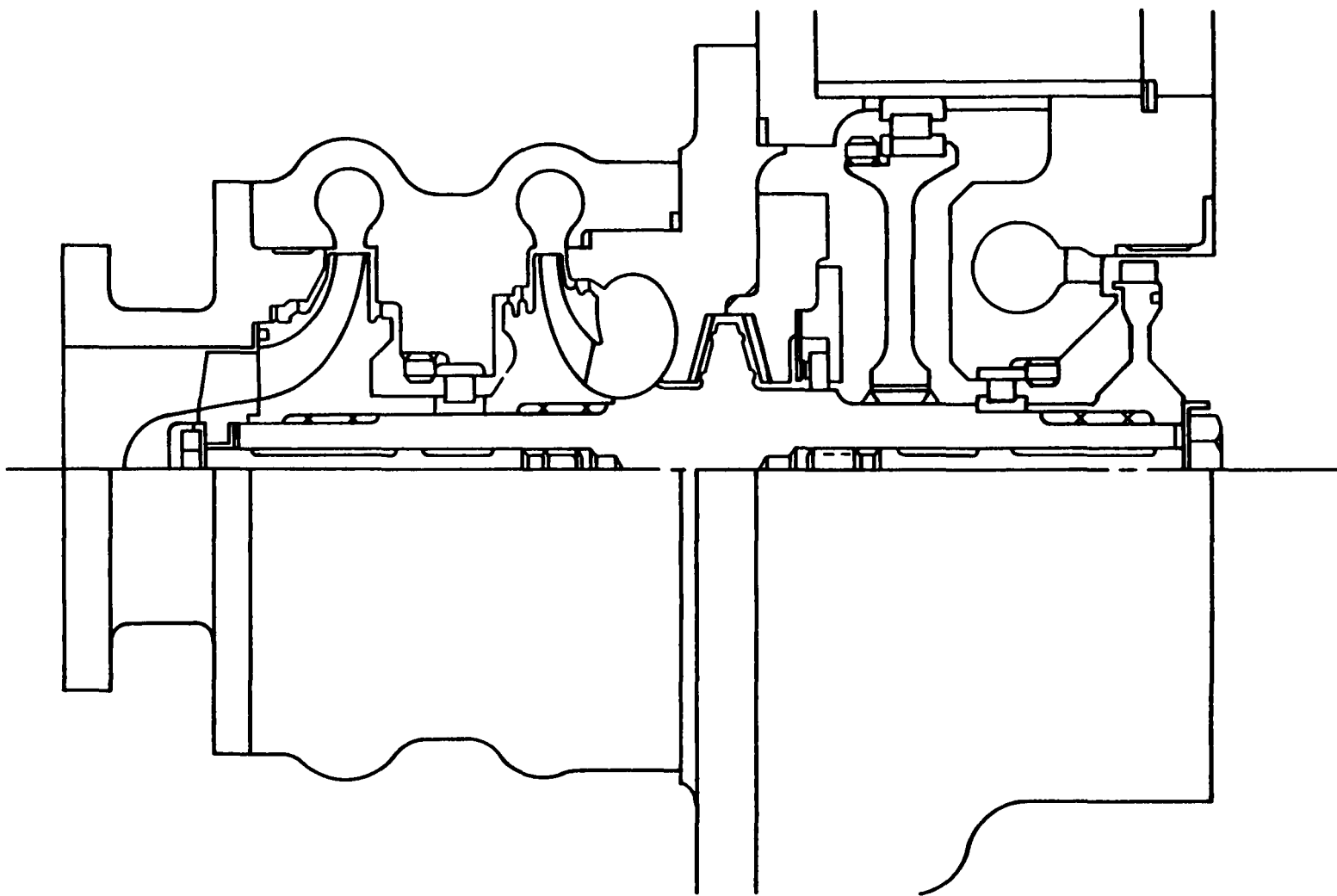


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Figure 8-16 — Fuel Pump Second-Stage Impeller Efficiency

TABLE 8-2 OTV IMPELLER RADIAL LOADS

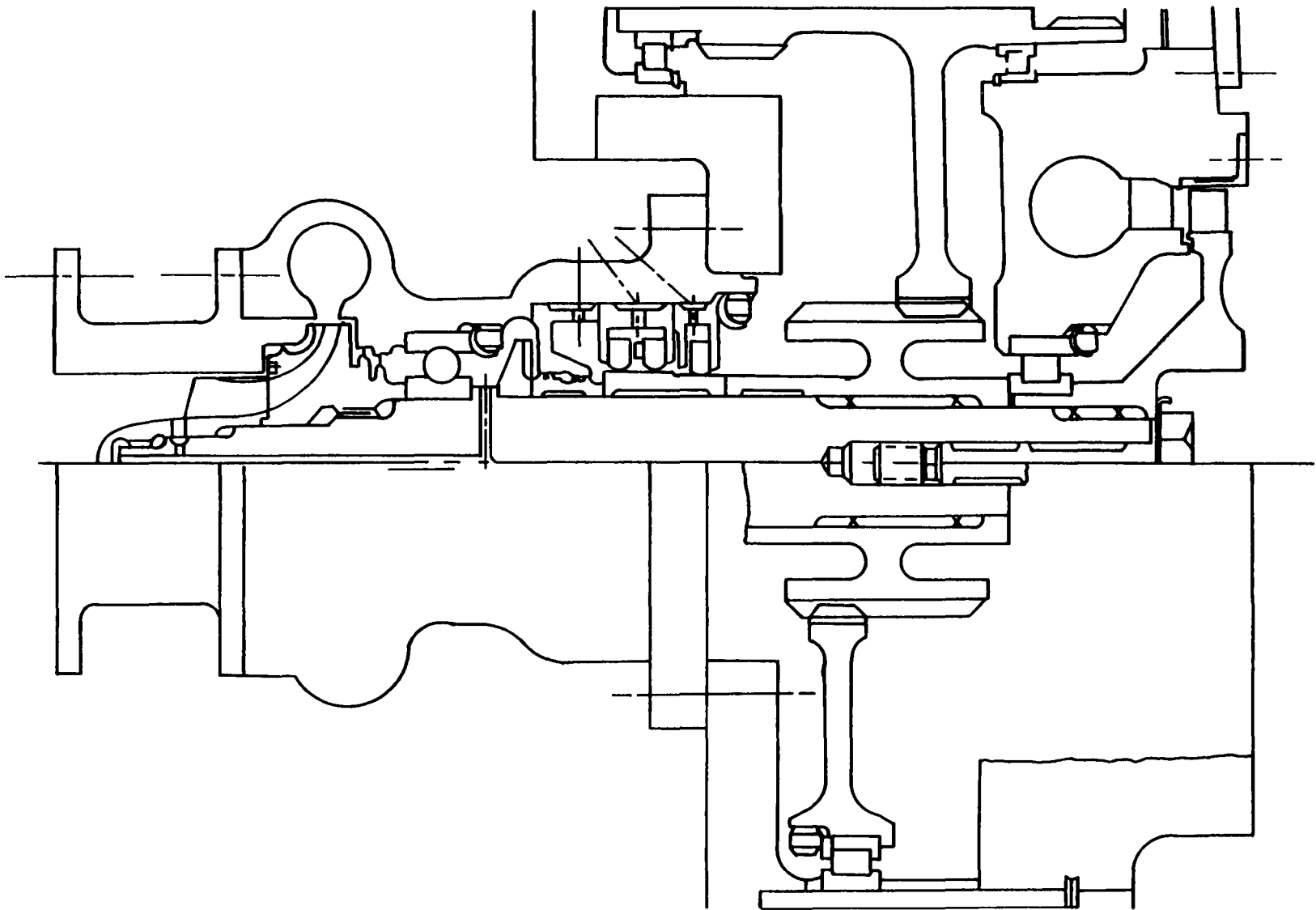
| Pump Location       | Pumped Idle            |        |                |    | Nominal Design          |         |                |    | Off Design              |         |                |    |
|---------------------|------------------------|--------|----------------|----|-------------------------|---------|----------------|----|-------------------------|---------|----------------|----|
|                     | O/F = 6.0 — 10% Thrust |        |                |    | O/F = 6.0 — 100% Thrust |         |                |    | O/F = 7.0 — 111% Thrust |         |                |    |
|                     | Q/N                    |        |                |    | Q/N                     |         |                |    | Q/N                     |         |                |    |
|                     | Q/N <sub>BEP</sub>     | N      | F <sub>T</sub> | No | Q/N <sub>BEP</sub>      | N       | F <sub>T</sub> | No | Q/N <sub>BEP</sub>      | N       | F <sub>T</sub> | No |
| Main LOX Pump       | 0.4                    | 17,631 | 47.6           |    | 1.0                     | 67,390  | 46.2           |    | 1.13                    | 68,709  | 212.4          |    |
| Fuel Pump 1st-Stage | 0.4                    | 39,244 | 47.1           |    | 1.0                     | 150,000 | 61.7           |    | 0.97                    | 152,932 | 98.2           |    |
| Fuel Pump 2nd-Stage | 0.4                    | 39,244 | 27.4           |    | 1.0                     | 150,000 | 71.6           |    | 0.97                    | 152,932 | 88.9           |    |



FD 212871

*Figure 8-17 — Main Fuel Pump*

A configuration sketch of the OTV main LOX pump and turbine drive is shown in Figure 8-18



FD 212871

Figure 8-18 — Main Oxidizer Pump

#### 8.5.4 Fuel Boost Pump

The fuel boost pump is an unshrouded axial flow, low speed inducer LSI type pump, designed for an inlet NPSH of 15 and 90 ft of thermodynamic suppression head (TSH). The fuel LSI is scaled from the P&WA Seajet 12-1V design to take advantage of its proven test performance. The design configuration employs three full length cambered blades and three splitter blades followed by a single discharge constant velocity volute collector with conical diffuser. The shaft is supported by a 10 mm roller bearing at the front end of the pump and by a 25 mm ball bearing at the back end.

The suction capability of 30,000  $N_{ss}$  for the fuel LSI design is at a design inlet tip flow coefficient of 0.11. Based on demonstrated Seajet 12-1V performance and collector loss calculations the overall efficiency of the fuel LSI is estimated to be approximately 75 percent, with a resultant shaft horsepower requirement of 25.3. The pump assembly is shown in Figure 8-19.

#### 8.5.5 Oxidizer Boost Pump

The oxidizer boost pump (LOX LSI) is a scaled version of the fuel LSI since it also has been scaled from the Seajet 12-1V design to take full advantage of its proven test performance. The LOX LSI is a larger scale design than the fuel LSI to provide the required flowrate and headrise with 2 ft of inlet NPSH and 5 ft of TSH. The LOX LSI has the same configuration features as the fuel LSI, employing three full-length cambered blades plus three cambered splitter blades, and a single discharge, constant velocity volute collector with conical diffuser. The shaft is supported by a 15 mm roller bearing at the front end of the pump and by a 25 mm ball bearing at the back end.

The 30,000  $N_{ss}$  suction capability of the LOX LSI is at its optimum design inlet tip flow coefficient of 0.11. As with fuel LSI design, the overall pump efficiency is estimated to be approximately 75 percent, establishing a shaft horsepower requirement of 10.1. A schematic of the pump assembly is shown in Figure 8-20.

#### 8.5.6 Fuel Turbine

The fuel pump drive turbine is an axial flow, full admission, single-stage design deriving its power from the expansion of the heated hydrogen propellant used to cool the thrust chamber/nozzle. A low reaction blade design was chosen to minimize axial thrust loads.

The fuel turbine develops 1630 hp at the design point with a total to static pressure ratio of 1.562 and an efficiency of 71.2 percent. The turbine design point is at 100 percent thrust and mixture ratio of 6.1. A comparison of the AEE fuel and oxidizer turbine efficiencies with past P&WA designs is shown in Figure 8-21, indicating the design efficiencies are consistent with previously achieved levels.

The fuel turbine elevation is shown in Figure 8-22. The axial chord lengths for the vanes and blades were set at 0.29 in. and 0.30 in., respectively. These were selected as being the smallest allowable to minimize aerodynamic losses of each airfoil. For a given height, small chords yield high aspect ratios, which in turn maintain low airfoil end losses. A blade radial tip clearance of 0.01 in. is required to achieve the design efficiency for the fuel turbine.

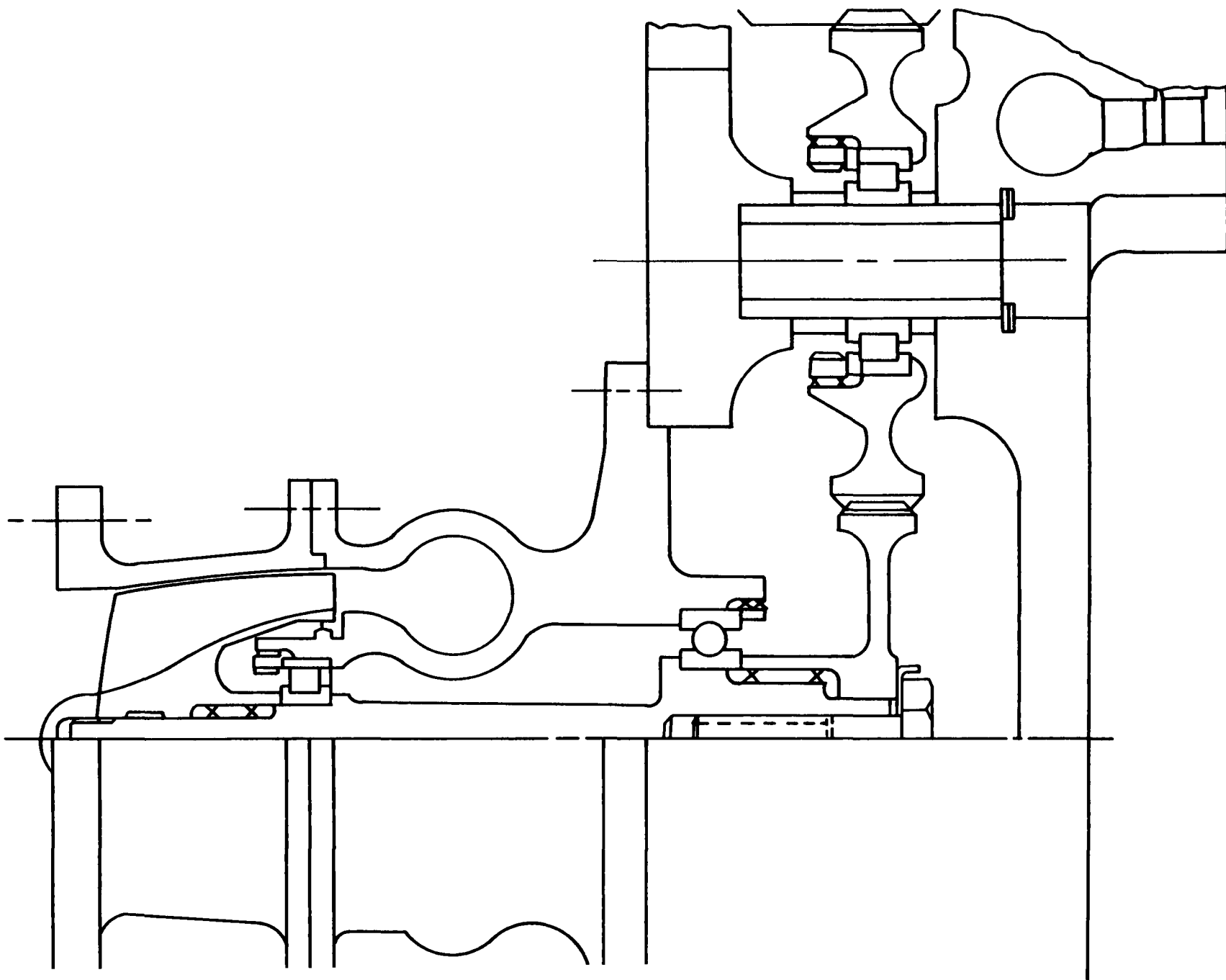
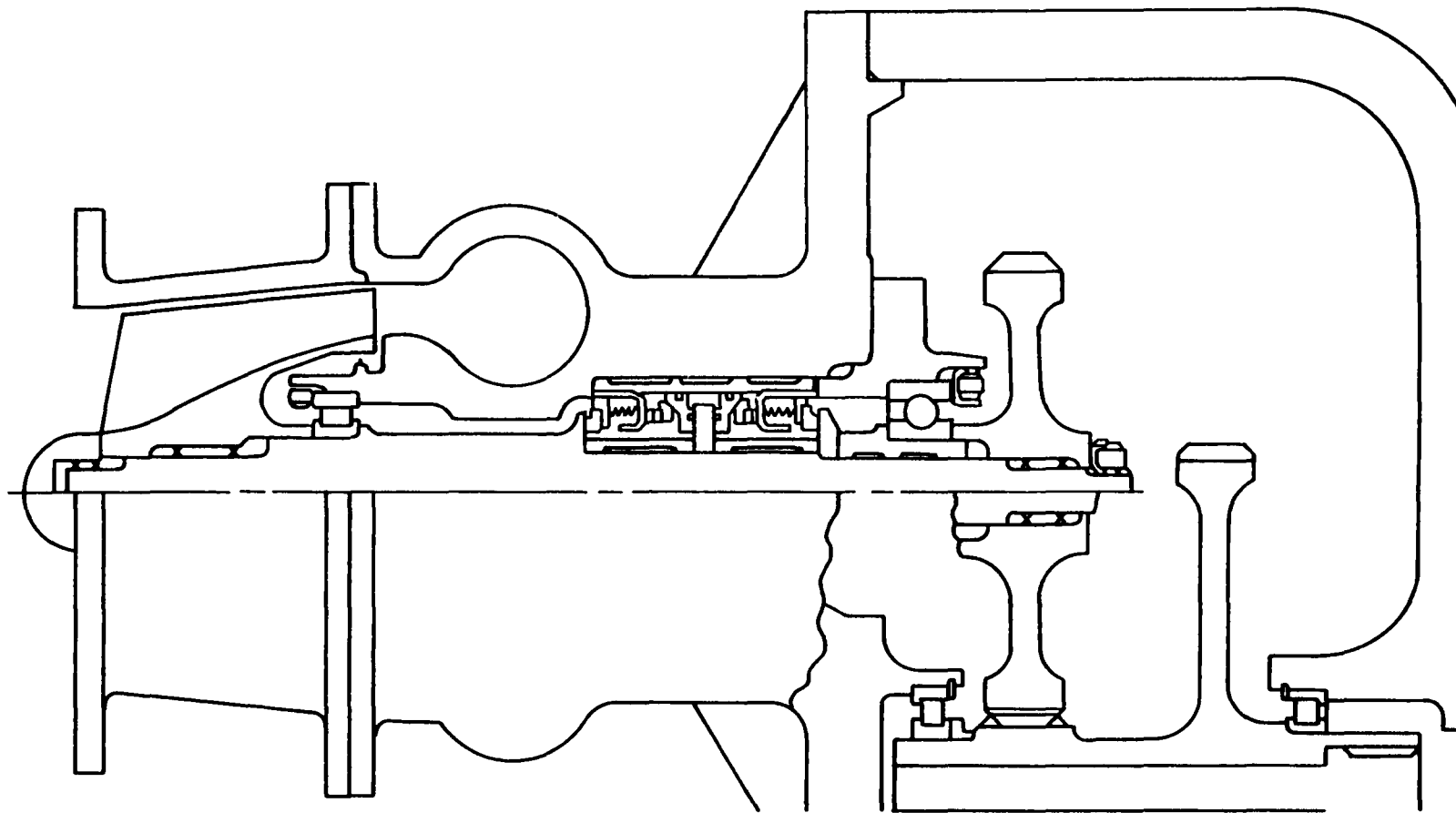
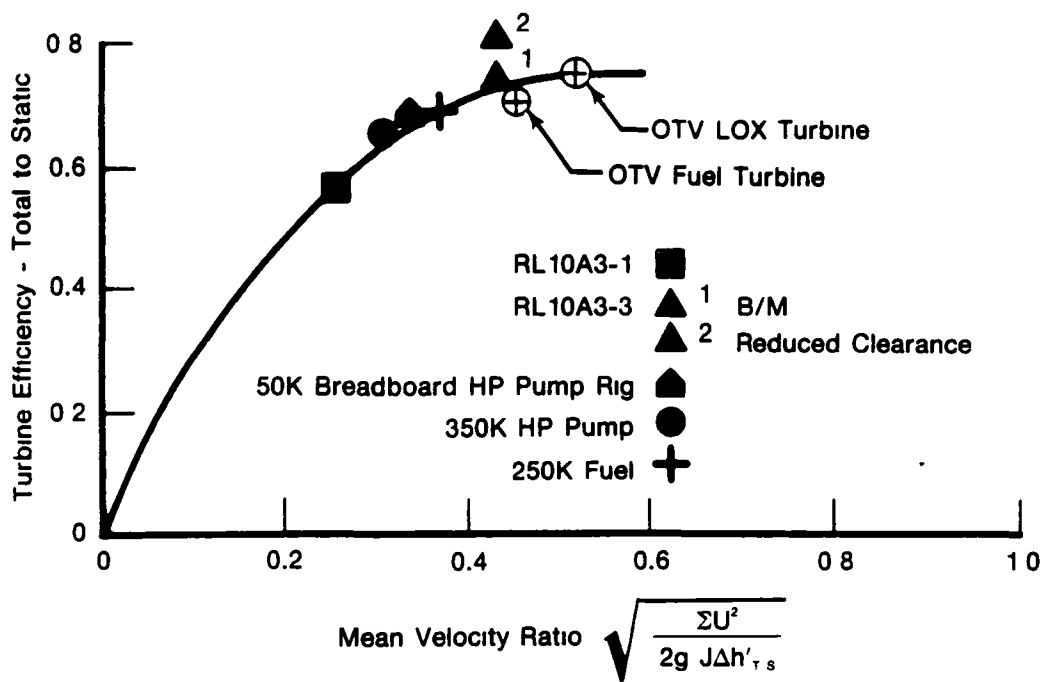


Figure 8-19 — Fuel LSI Pump



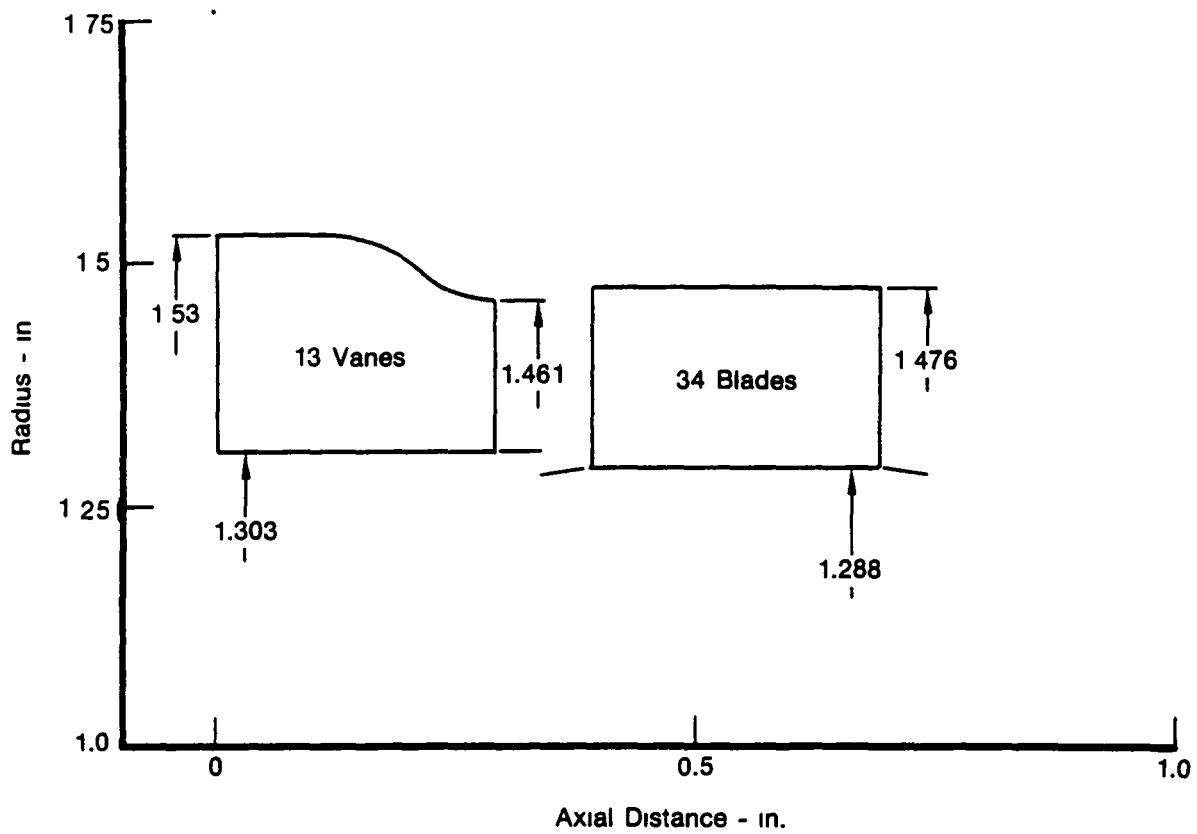
FD 212873

Figure 8-20 — Oxidizer LSI Pump



FD 212877

Figure 8-21 — OTV Engine Turbine Efficiencies



FD 212875

Figure 8-22 — Fuel Turbine Elevation

### 8.5.7 Oxidizer Turbine

The oxidizer pump drive turbine is an axial flow, full admission, single-stage design placed in series with the fuel turbine, therefore utilizing the same driving fluid. This turbine is also used to drive the fuel and oxidizer low speed inducers through a gearing system. As with the fuel turbine, a low reaction blade design was chosen to minimize axial thrust loads.

A comparison of the AEE fuel and oxidizer turbine efficiencies with past P&WA designs is shown in Figure 8-21, indicating the design efficiencies are consistent with previously achieved levels.

The oxidizer turbine design elevation schematic is shown in Figure 8-23. The oxidizer turbine requires a blade radial tip clearance of 0.01 in. to achieve the design efficiency.

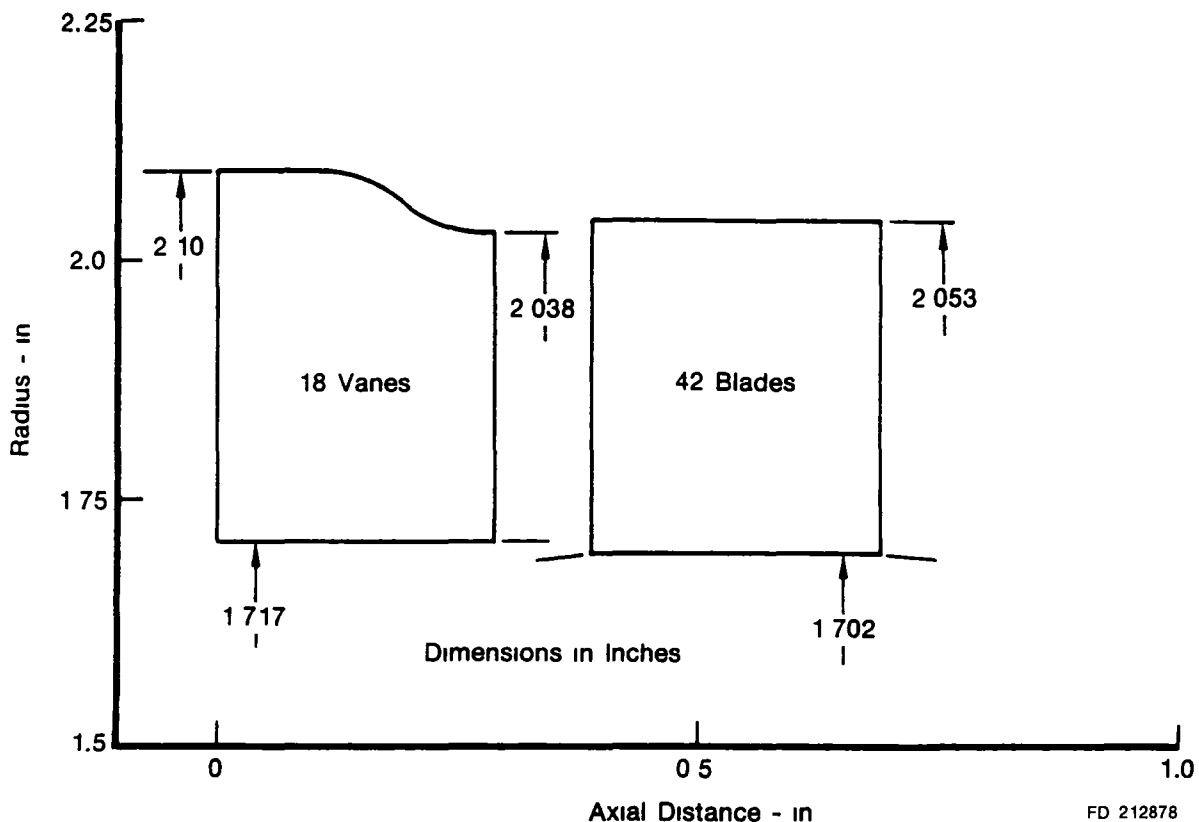


Figure 8-23 — Oxidizer Turbine Elevation

### 8.5.8 Bearings and Main Shafts

Very high rotational speeds for the turbopump produce an increased concern over bearing loads and turbomachinery shaft critical speed margin.

The performance goals of the OTV engine necessitate high technology components. The resulting turbomachinery contains lightweight, high-speed rotors. Table 8-3 summarizes the various pump rotor weights and maximum operating speeds. Rotor dynamics analyses of the main fuel pump, main oxidizer pump, fuel low speed inducer, and oxidizer low speed inducer

predict acceptable margins over the maximum operating speeds as shown in Figure 8-24. For this design, acceptable critical speed margins were defined as a 15 percent margin over maximum operating speed for modes with less than 25 percent rotor strain energy and a 25 percent margin over maximum operating speed for modes with more than 25 percent rotor strain energy. These criteria require careful rotor design and multiplane balancing during final pump assembly.

TABLE 8-3 — TURBOMACHINERY ROTOR WEIGHTS AND OPERATING SPEEDS

| <i>Pump</i>                | <i>Rotor Weight<br/>lb</i> | <i>Maximum Operating<br/>Speed (rpm)</i> |
|----------------------------|----------------------------|--|
| Main Fuel Pump             | 2.24                       | 150,000                                  |
| Main Oxidizer Pump         | 2.98                       | 67,390                                   |
| Fuel Low Speed Inducer     | 1.90                       | 46,021                                   |
| Oxidizer Low Speed Inducer | 2.16                       | 8,850                                    |

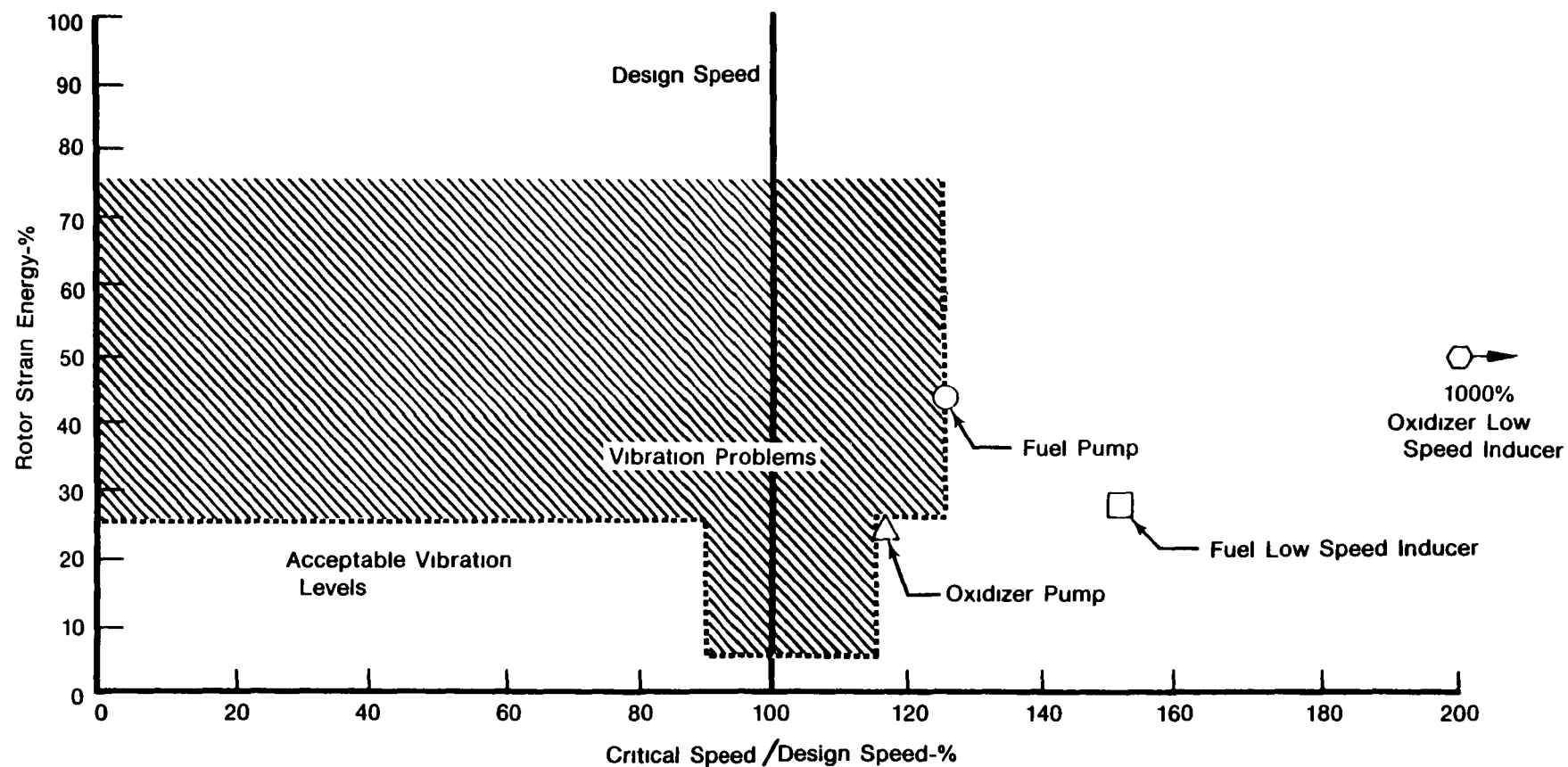
The turbopump ball and roller bearings were evaluated using Jones II Bearing Analysis Deck (A926). A life factor of one (1X) was used for both ball and roller bearings. The 100 hr design life includes a 10 percent reliability factor used for cryogenic applications. The dynamic bearing loads (Figures 8-25 and 8-26) were calculated by assuming unbalances equal to the weight of the impellers and turbine disks offset 0.001 in. from the rotor centerline, with phase relationships that produce the maximum bearing load. The analysis revealed that the fuel pump roller bearings will not reach a life of 100 hr unless silicon nitride elements are used instead of steel elements. The use and manufacture of silicon nitride roller bearings was demonstrated during a high speed roller bearing test program conducted at Orenda Ltd. in 1973. A bore reduction to 18 mm from 20 mm would provide the 100 hr failure life for the silicon nitride roller bearings at 150,000 rpm for the fuel pump roller bearings. Decreasing the pump speed to 147,000 rpm will also provide the required 100 hr fatigue life for 20 mm bearings. Since the design point operating speed of the fuel turbopump is 147,000 rpm, the life requirement was met although with no excessive margin.

Conventional shaft materials used in the design analysis ( $E = 31 \times 10^6$ ,  $\rho = 0.298 \text{ lb/in}^3$ ). A 20 percent improvement in shaft materials has been demonstrated in a recent material technology program. This improved shaft material will be used in the advanced OTV engine turbopump producing improvement in critical speed margins over that discussed above as well as a decreased bearing load.

## 8.6 ENGINE CONTROL VALVES

The location of the advanced OTV engine valves is shown in Figure 8-27. Several of these valves are similar to the ones used in the RL10 engine (i.e., the propellant inlet shutoff valves, main fuel shutoff valve, and the solenoid valves).

The oxidizer flow control valve (Figure 8-28) is a spring-loaded, normally closed, line pressure actuated valve. It is configured to provide ground trim of full thrust propellant mixture ratio. The valve contains a spring-loaded poppet valve used to meter oxidizer flow during full thrust and regulate flow during the engine transient to full thrust. The poppet valve is spring-loaded closed and opens as a function of the pressure differential between valve inlet pressure and a pressure within the valve cavity which has been vented to pump inlet pressure. During tank head idle and pumped idle operation, the poppet is closed and liquid oxidizer is not allowed to enter the injector. When the engine accelerates from pumped idle to full thrust operation, the main poppet valve is also opened as a function of the differential pressure between valve inlet and pump inlet pressure. The bypass and main poppet valves both remain open during full



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Figure 8-24 — OTV Pumps Critical Speed Analysis

thrust operation and the combined areas meter the required oxidizer flow. The full open position of the main poppet valve can be ground trimmed by a threaded mechanical stop to ground adjust engine mixture ratio.

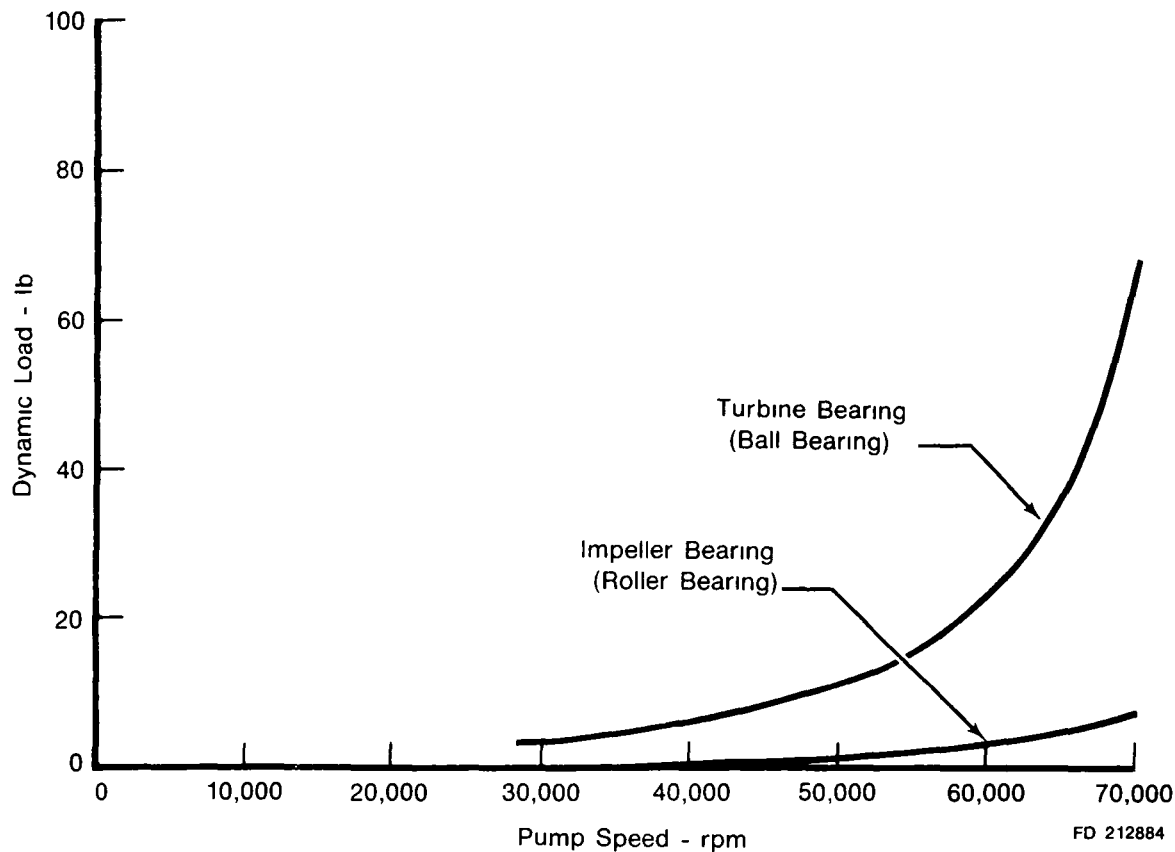
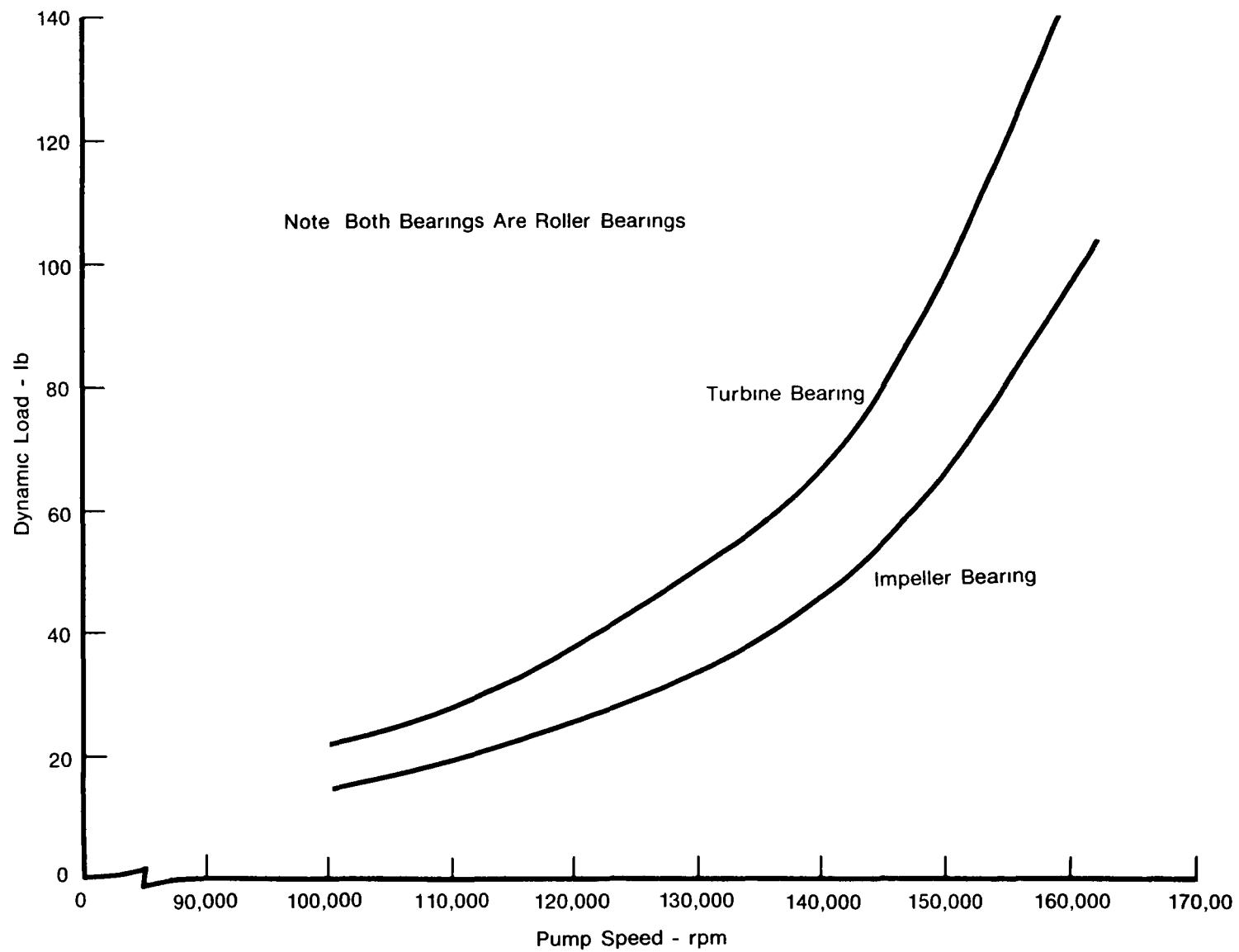


Figure 8-25 — Oxidizer Pump Dynamic Bearing Loads

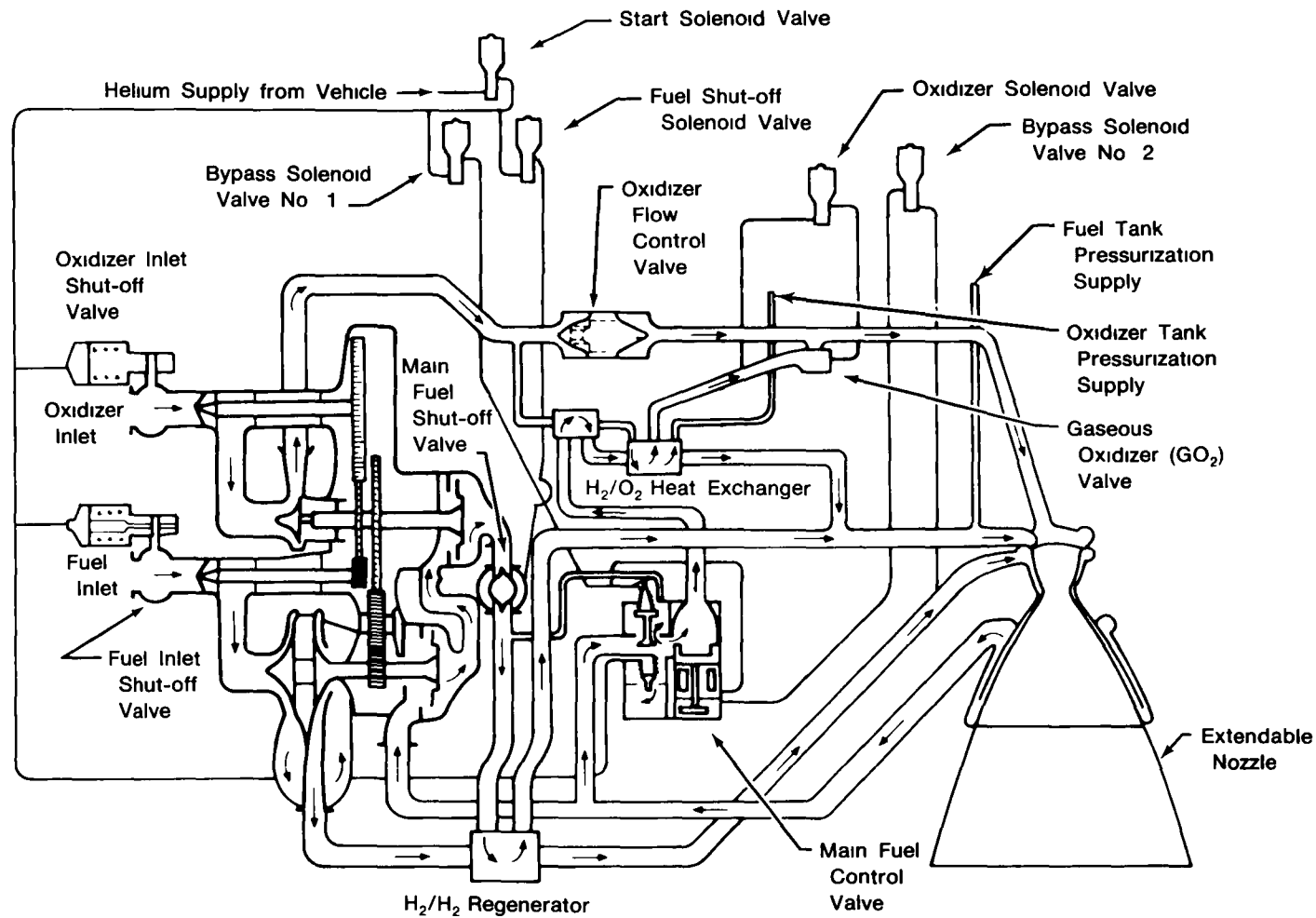
The propellant inlet shutoff valves (Figure 8-29) are spring loaded, normally closed, helium operated, two position ball valves that provide a seal between the vehicle propellant tank and the engine pumps. Both valves are located just upstream of their respective pump inlets and are of the same respective diameter as the fuel and oxidizer pump inlets. The valves are actuated by helium operating on a piston bellows assembly. The linear motion of the actuator is translated by rack and pinion into a rotary motion at the ball valve. Ball sealing is accomplished with dual pressure loaded fluorocarbon rub seals. The valves incorporate a vented cavity between the dual seals such that any leakage past the closed valve is vented overboard.

The main fuel shutoff valve (Figure 8-30) is a helium operated, two position, normally closed annular gate valve. The valve serves to prevent the flow of fuel through the fuel pump turbine during tank head idle operation and provides a rapid cutoff of fuel flow to the combustion chamber at engine shutdown. The shutoff gate is opened by helium pressurization of a bellows assembly to allow the flow of fuel through the turbine at the operating modes above tank head idle. The compressed shutoff valve spring returns the gate to its normally closed position when helium pressure is vented at engine shutdown. Sealing is accomplished by the sealing of the spherical surface of the gate seal ring.



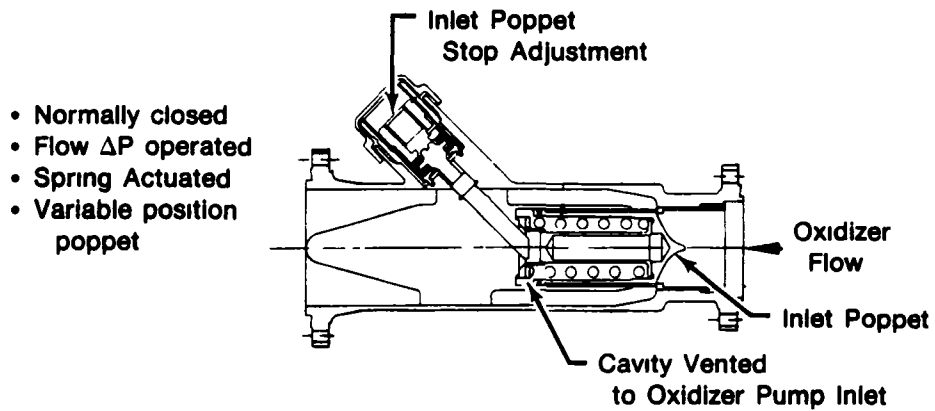
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Figure 8-26 — OTV Fuel Pump Dynamic Bearing Loads



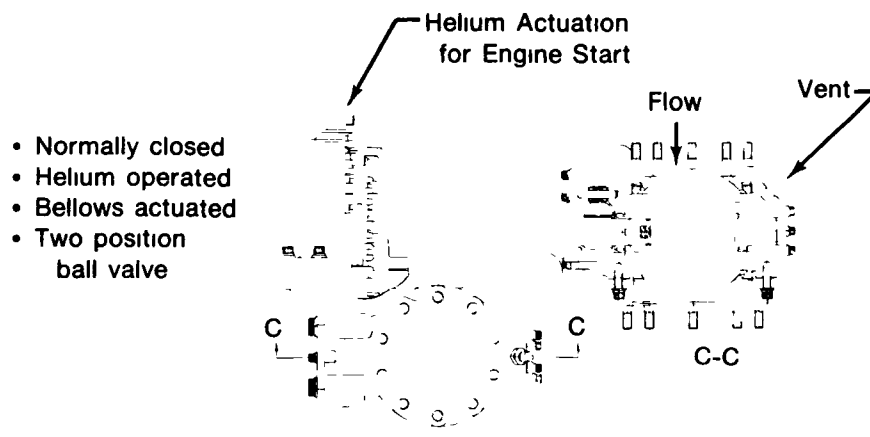
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Figure 8-27 — Engine Valve Location Schematic



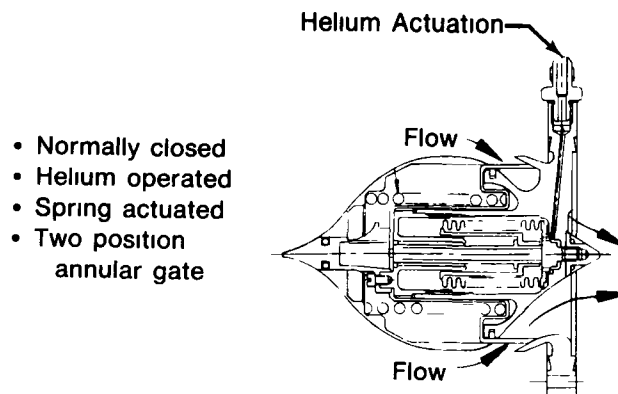
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Figure 8-28 — Oxidizer Flow Control Valve



FD 212894

Figure 8-29 — Fuel and Oxidizer Propellant Inlet Shutoff Valves



FD 212895

Figure 8-30 — Main Fuel Shutoff Valve

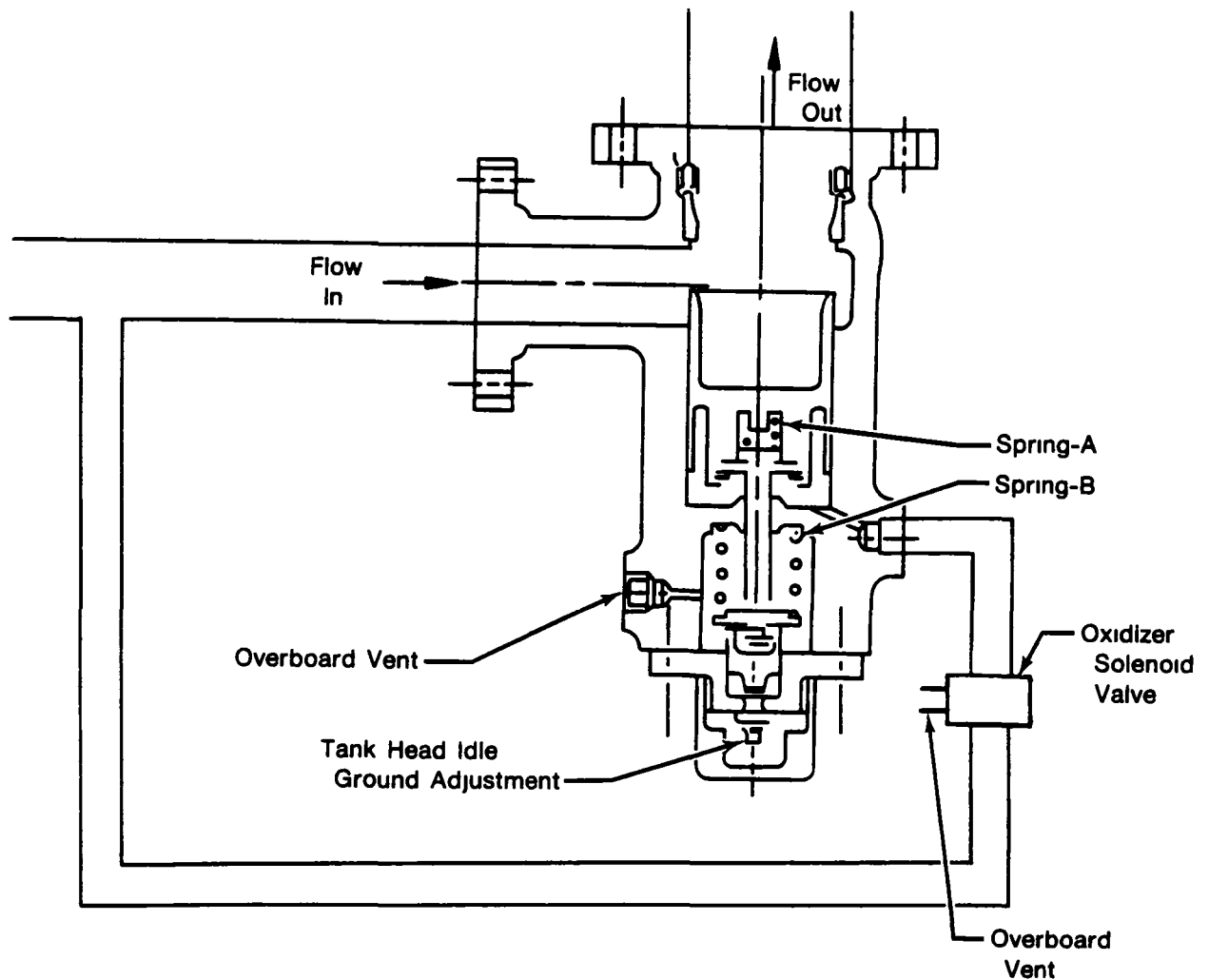
The gaseous oxidizer valve (Figure 8-31) is a spring loaded, normally open, pressure actuated sleeve valve located between the GOX heat exchanger and the injector. This valve meters gaseous oxygen flow during tank head and pumped idle operation and regulates oxygen flow during the transient to pumped idle. In the tank head idle mode, the valve is normally partially opened to a predetermined position to meet the required oxidizer flow. The sleeve valve is opened fully during the pumped idle mode by the increase in oxidizer line pressure acting on the face of the sleeve which compresses the spring within the sleeve/piston assembly. During the full thrust operation the valve is closed, actuated by the oxidizer pump discharge pressure acting on the piston of the sleeve/piston assembly.

The main fuel control valve (Figure 8-32) provides the control functions of turbine bypass flow for thrust regulation, ventage of fuel at shutdown, and provides fuel flow to hydrogen regenerator during tank head idle operation. The thrust control portion of the valve is a normally closed, helium and hydrogen pressure actuated, three position sleeve bypass valve used to control engine thrust by regulation of turbine power. Control of engine thrust is provided at full thrust by a ground adjusted needle valve which allows approximately 4 percent hydrogen flow to bypass the closed sleeve valve and therefore bypass the turbines. The valve is also pressure actuated to allow the setting of two discrete areas for metering turbine bypass hydrogen flow during tank head and pumped idle operation. During tank head idle operation, the valve is actuated to full open position by helium pressure action on the concentric (annular) piston assembly. During pumped idle operation the valve is actuated to an intermediate area by gaseous hydrogen acting on a secondary concentric piston as the annular helium piston is vented. Holes are provided through the valve's sleeve face to maintain hydrogen pressure on both sides of the face, in order to reduce the spring load required to move the valve from the full open to the intermediate position.

The fuel vent portion of the main fuel control is a pressure operated, two-position poppet valve that is spring loaded open to provide pressure relief of the fuel system lines during engine shutdown. The valve is maintained in the closed position during all three active modes of engine operation. At the start signal, helium pressure actuates the valve assembly, moving the valve to close the overboard vent port. At shutdown, when helium pressure is removed, the vent port opens fully relieving fuel pressure in the fuel system lines.

The hydrogen regenerator flow portion of the main fuel control is a pressure operated, two-position, poppet valve that is spring loaded, normally closed. At the start signal for tank head idle operation, helium pressure actuates the valve assembly, moving the poppet to the full open position thereby providing hydrogen flow to the hydrogen regenerator. The valve is maintained in the closed position for the pumped idle and full thrust modes of engine operation.

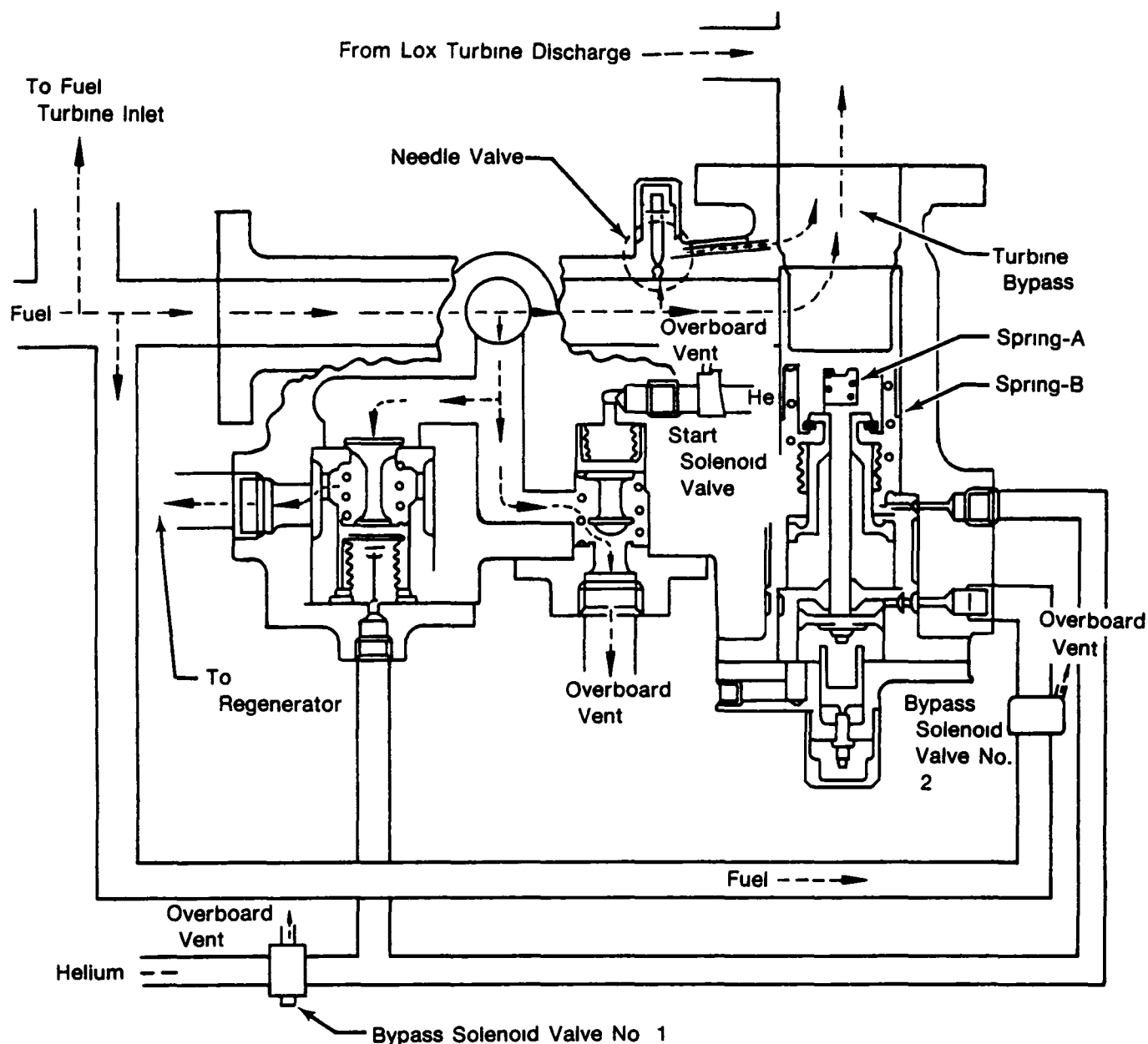
The solenoid valves (Figure 8-33) are solenoid actuated, direct acting, three-way valves with double-ended poppets that supply helium, hydrogen or oxygen actuation pressure to the various propellant valves. The five solenoid valves used in the OTV engine are identical in design and function. The start solenoid valve controls the actuator helium supply to the fuel shutoff valve. Bypass solenoid valve No. 1 controls the actuator helium supply to the turbine bypass valve and hydrogen regenerator flow valve, both on the main fuel control, for tank head idle operation. Bypass solenoid No. 2 controls the actuator hydrogen supply to the turbine bypass valve for pumped idle operation. The oxidizer solenoid valve controls the actuator oxidizer supply to the gaseous oxidizer valve, for full thrust operation.



- 1 Tank Head Idle Setting (as drawn) Solenoid valve is closed Valve flow path pressure insufficient to compress spring-A
- 2 Pumped Idle Setting Solenoid valve is closed Valve flow path pressure sufficient to compress spring-A (valve opens farther than shown)
- 3 Transient from Pumped Idle to Full Thrust Solenoid valve opens As valve pressure increases, valve closes (Function of spring-B spring rate) Valve is fully closed above thrust levels of about 30 percent of rated thrust

FD 212896

Figure 8-31 — Gaseous Oxidizer Valve

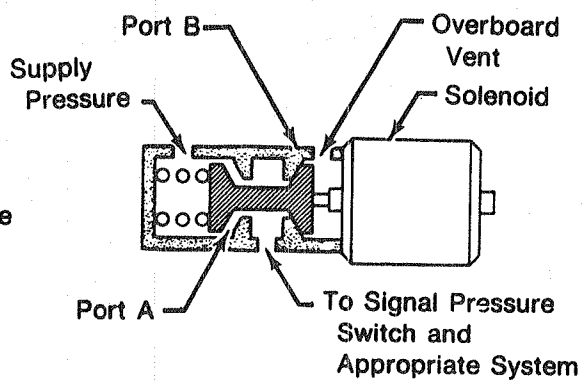


|                |  |
|----------------|--|
| Off            | All solenoid valves closed. Valve in positions shown Turbine bypass through needle valve only No flow to the regenerator. Overboard vent valve open (Fail safe position)   |
| Tank Head Idle | Bypass solenoid valve No. 1 open, No. 2 closed. Start solenoid valve open. Turbine bypass open (springs A and B compressed) Flow to regenerator. Overboard vent valve closed   |
| Pumped Idle    | Bypass solenoid valve No. 1 closed, No. 2 open Start solenoid valve open. Turbine bypass is a function of valve inlet pressure and spring-B (spring-A is not compressed) No flow to regenerator Overboard vent valve closed. |
| Full Thrust    | Bypass solenoid valves No. 1 and No. 2 closed. Start solenoid valve open Turbine bypass through needle valve only. No flow to regenerator Overboard vent valve closed  |

Figure 8-32 — Main Fuel Valve

FD 212897

- Normally closed
- Electrically operated
- Spring actuated
- Two position - double ended poppet



FD 212898

Figure 8-33. — Solenoid Valves

## SECTION 9.0 CONCLUSIONS AND RECOMMENDATIONS

This study program has shown that an advanced expander cycle engine (AEE) can be designed which can provide high performance for an OTV application. The critical components of this engine are the thrust chamber, the turbomachinery, the extendible nozzle system, and the engine throttling system.

In order to proceed with confidence into a full-scale development program for such an engine, it is recommended that these critical engine areas be addressed in a future technology program. This technology program has been arranged in three general phases. These phases are of increasingly focused activity beginning from basic material evaluations and ending with the fabrication and test of major component assemblies as shown in Table 9-1.

TABLE 9-1 — FOLLOW-ON TECHNOLOGY PROGRAM AREAS

| <i>Component</i>  | <i>Phase I<br/>(Material and<br/>System Selections)</i>   | <i>Phase II<br/>(Subcomponent<br/>Evaluations)</i>   | <i>Phase III<br/>(Component<br/>Tests)</i>   |
|---|---|--|--|
| A Thrust chamber <ul style="list-style-type: none"> <li>• Performance</li> <li>• Life</li> <li>• Fabrication</li> </ul> | <ul style="list-style-type: none"> <li>• Copper alloy properties evaluation</li> </ul>                        | <ul style="list-style-type: none"> <li>• Scale model LCF tests</li> <li>• Chamber segment fab</li> </ul>               | <ul style="list-style-type: none"> <li>• Thrust chamber rig test</li> </ul>                          |
| B Turbomachinery <ul style="list-style-type: none"> <li>• Dynamics</li> <li>• Life</li> </ul>                           | <ul style="list-style-type: none"> <li>• Material properties evaluation for gears, bearings, seals</li> </ul> | <ul style="list-style-type: none"> <li>• Gear rig test</li> <li>• Bearing rig test</li> <li>• Seal rig test</li> </ul> | <ul style="list-style-type: none"> <li>• Turbopump rig test</li> </ul>                               |
| C Extendible nozzle <ul style="list-style-type: none"> <li>• Weight</li> <li>• Life</li> </ul>                          | <ul style="list-style-type: none"> <li>• Nozzle material evaluation</li> </ul>                                | <ul style="list-style-type: none"> <li>• Actuation mechanism design and test</li> </ul>                                | <ul style="list-style-type: none"> <li>• Stub nozzle hot firing</li> </ul>                           |
| D Engine throttling system  | <ul style="list-style-type: none"> <li>• System selection/design</li> </ul>                                   |  | <ul style="list-style-type: none"> <li>• Injector rig test (combined with thrust chamber)</li> </ul> |

To assure that the technology program is properly focused to meet the evolving requirements of the OTV program, a continuing coordination activity between the engine contractor, the vehicle systems contractors, and the mission planners is needed.

This activity will encompass

- Providing updated engine characteristics for OTV program analysis
- Evaluating the impact of OTV program decisions
  - Aeroassisted or all-propulsive OTV (or mixed)
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  - Thrust levels (or full throttling) requirements
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